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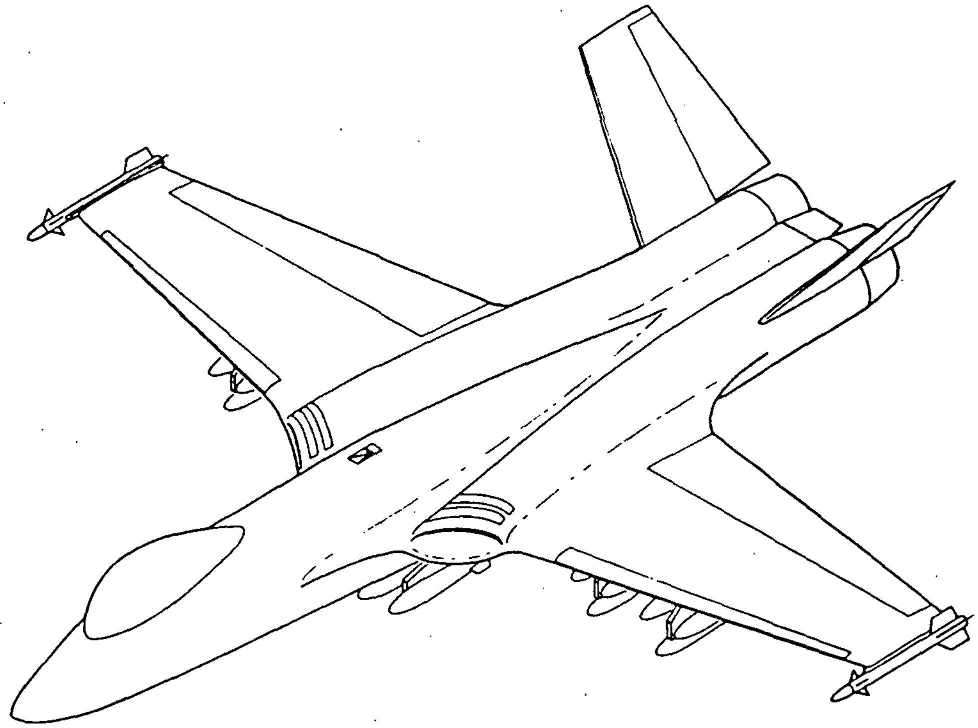
PROPOSAL FOR A

CLOSE AIR SUPPORT

AIRCRAFT:

CYCLONE

A CLOSE AIR SUPPORT AIRCRAFT FOR TOMORROW



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ABSTRACT

To meet the threat on the battlefield of the future, the U.S. ground forces will require reliable air support. To provide this support future aircrews demand a versatile close air support aircraft capable of delivering ordinance during the day, night or in adverse weather with pin-point accuracy. The Cyclone aircraft meets these requirements, packing the 'punch' necessary to clear the way for effective ground operations. Possessing anti-armor, missile and precision bombing capability, the Cyclone will counter the threat into the twenty-first century. This proposal shows the Cyclone to be the realistic economical answer to the demand for a capable close air support aircraft. The Cyclone is not designed to make life hard on the enemy; it is designed to make it very short.

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF TABLES

SYMBOLS

A	Area
A_{cap}	Capture Area
AR	Aspect Ratio
b	Wing Span
c	Mean Aerodynamic Chord
C_D	Aircraft Drag Coefficient
C_{D_b}	Base Drag Coefficient
c_{dc}	Crossflow Drag Coefficient
C_{D_i}	Induced Drag Coefficient
C_{D_L}	Drag-due-to-lift Coefficient
C_{D_o}	Zero Lift Drag Coefficient
c_f	Friction Coefficient
$C_{f/c}$	Chord Ratio of Flaps to Wing
C_L	Aircraft Lift Coefficient
C_{L_h}	Horizontal Tail Lift Coefficient
C_{l_α}	Airfoil Lift Slope
C_{L_α}	Lift Due to Angle of Attack Derivative
C_{L_o}	Lift Coefficient for Zero Angle of Attack
c_r	Root Chord
c_t	Tip Chord
C_{y_β}	Side Force Due to Yaw Derivative
d	Diameter
D	Drag
d_b	Equivalent Diameter
$d\varepsilon/d\alpha$	Downwash Gradient of Tail
e	Wing Efficiency
E	Endurance
E_t	Touchdown Energy
f	Friction Factor
h	Height
h_h	Height of Horizontal Tail Above Wing Chord
HUD	Head Up Display
I	Moment of Inertia
INS	Inertial Navigation System

Δ	Increment
Δ_{cl}	Incremental Lift Due to Flaps
$\Delta_{\epsilon f}$	Change in Tail Downwash Due to Flaps
δ	Deflection
δ_f	Flap Deflection
ε	Downwash Angle of Horizontal Tail
Γ_w	Wing Dihedral
η_h	Ratio of Dynamic Pressures at Wing and Tail
Λ	Sweep Angle
$\Lambda_{c/4}$	Quarter Chord Sweep
Λ_{le}	Leading Edge Sweep
λ	Taper Ratio
μ_{inl}	Area Ratio, Inlet
μ_r	Rolling Friction Coefficient
σ	Stress
ω_n	Natural Frequency
ξ_p	Damping Ratio

Subscripts

av	Available
c/4	Quarter Chord
cg	Center of Gravity
cr	Cruise
dyn	Dynamic Load
el	Electrical
extr	Extraction
fus	Fuselage
h	Horizontal Tail
le	Leading Edge
mech	Mechanical
pneum	Pneumatic
requ	Required
sp	Short Period
to	Take Off
tot	Total
wf	Wing-Fuselage

INS	Inertial Navigation System
K_A	Downwash Coefficient
K_{cw}	Interference Factor
K_h	Downwash Coefficient
K_λ	Downwash Coefficient
K_{wh}	Interference Factor
L'	Airfoil Thickness Location Parameter
LANTIRN	Low Altitude Navigation and Targeting by Infra-Red at Night
L/D	Lift to Drag Ratio
l_m	Main Gear Distance to C.G.
l_n	Nose Gear Distance to C.G.
M	Mach Number
m_a	Engine Massflow
M_c	Crossflow Mach Number
M_i	Bending Moment
M_q	Dimensional Variation of Pitching Moment with Pitch Rate
M_u	Dimensional Variation of Pitching Moment with Speed
n	Load Factor
$n_{inl/inc}$	Incompressible Inlet Efficiency
P	Power
P_A	Power Available
P_{extr}	Power Extraction
Per	Perimeter
P_m	Main Gear Load
P_n	Nose Gear Load
P_{req}	Power Required
P_s	Specific Excess Power
P_{tot}	Total Pressure
q_∞	Dynamic Pressure
R	Range
R	Reynolds Number
RFP	Request For Proposal
R_{ls}	Lifting Surface Correction Factor
ROC	Rate of Climb

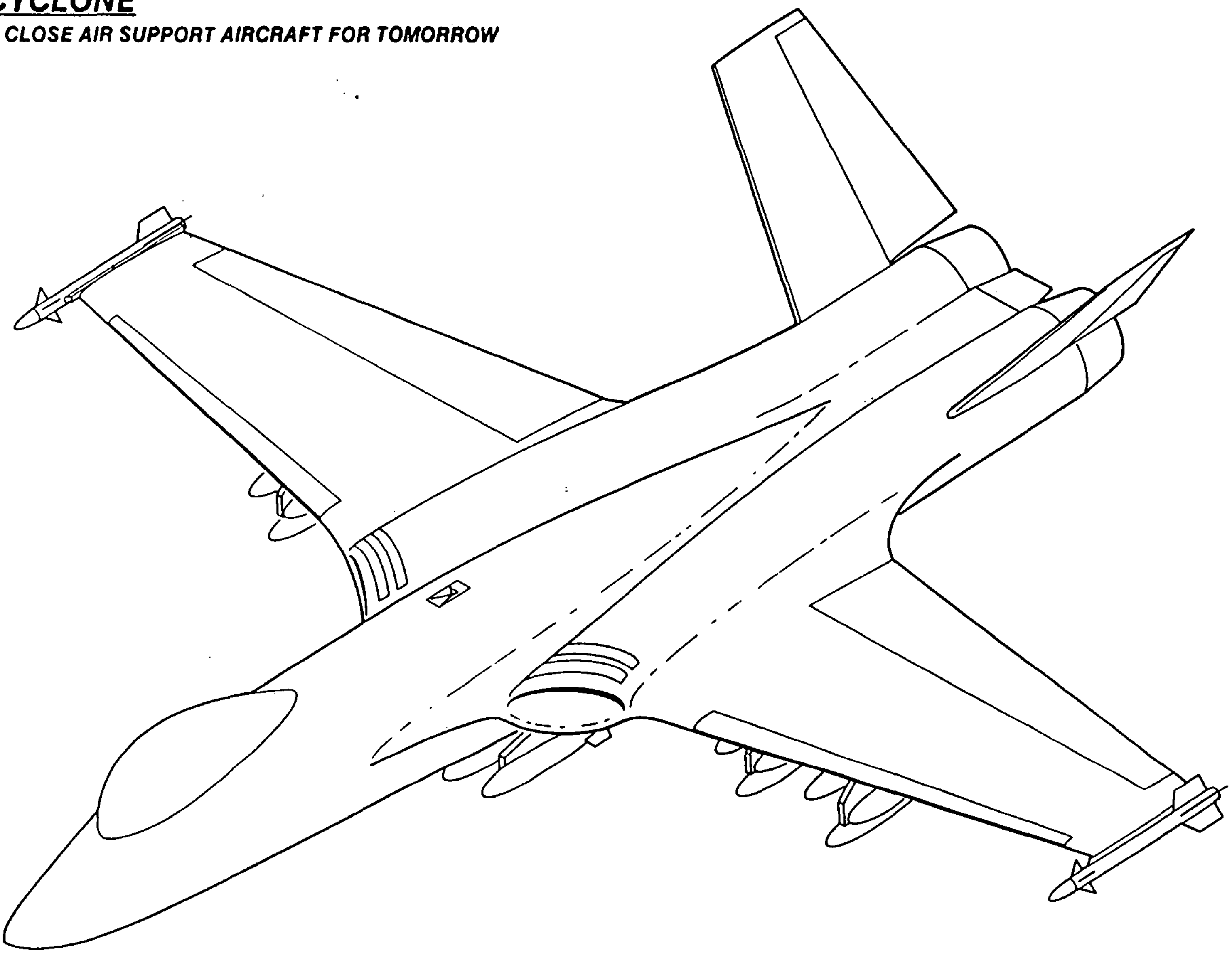
R_{wf}	Wing-fuselage Interference Factor
S	Wing Planform Area
S_{bfus}	Fuselage Base Area
S_{can}	Canopy Area
SHP	Shaft Horsepower
S_{LO}	Lift Off Distance
S_{plf}	Fuselage Planform Area
S_{wet}	Wetted Area
T_A	Thrust Available
t/c	Thickness to Chord Ratio
T_{requ}	Thrust Required
T/W	Thrust to Weight Ratio
U	Velocity
U_{de}	Gust Velocity
V	Airspeed
V_i	Shear
V_T	Touchdown Speed
V_{WF}	Wing Volume
W	Weight
W_{to}	Take Off Weight
X	Distance
x_q	Dimensional Variation of X_s Force with Speed
x_u	Dimensional Variation of X_s with Speed
y_u	Dimensional Variation of Y_s with Speed
z_h	Height of Horizontal Tail from Wing Chord
z_q	Dimensional Variation of Z_s Force with Speed

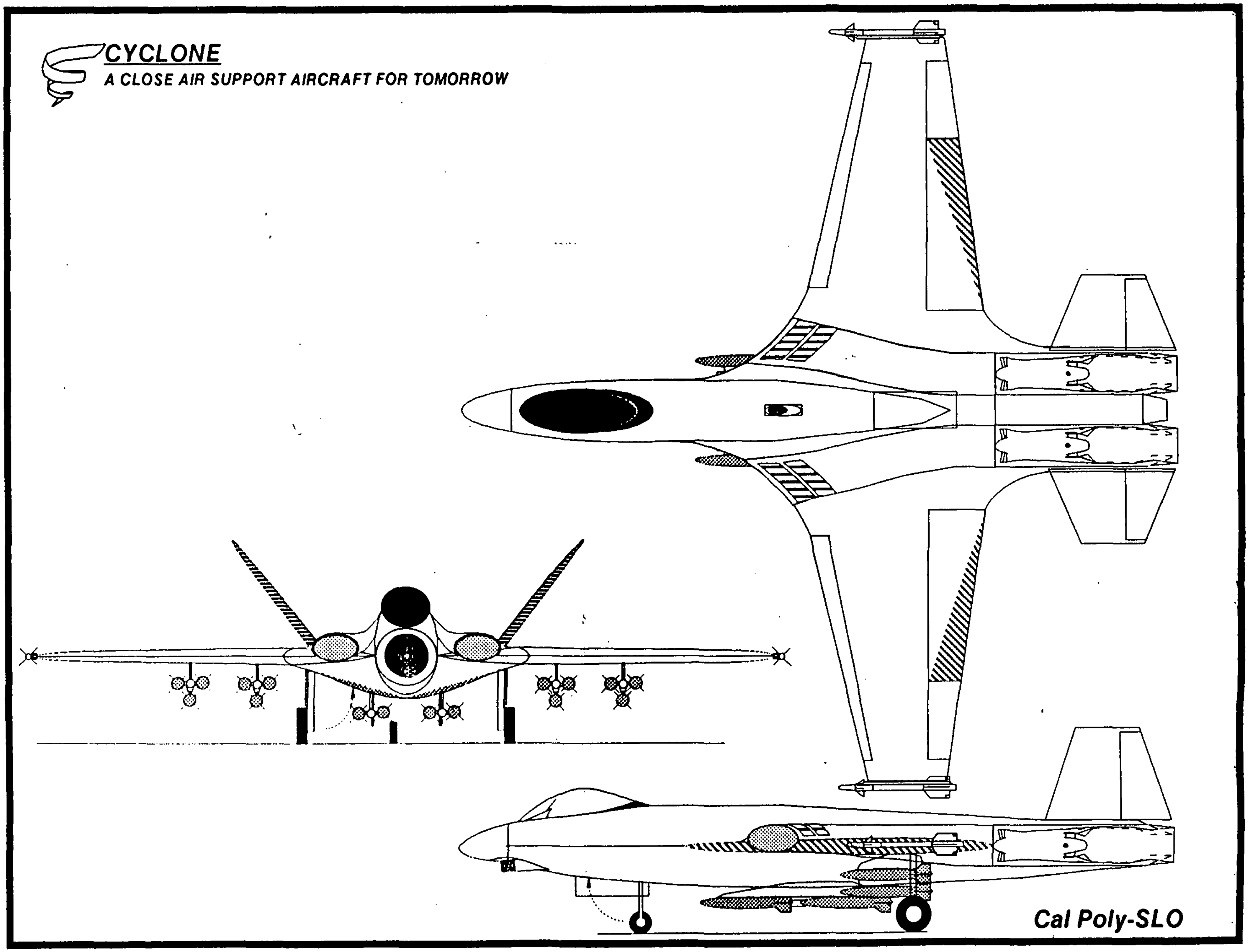
Greek Symbols

α	Angle of Attack
α_{CLmax}	Maximum Lift Angle of Attack
α_δ	Airfoil Lift Effectiveness Parameter
α_{ol}	Airfoil Zero Lift Angle of Attack
α_{oL}	Aircraft Zero Lift Angle of Attack
α_{stall}	Stall Angle of Attack
β	Equation Coefficient

CYCLONE

A CLOSE AIR SUPPORT AIRCRAFT FOR TOMORROW




CYCLONE
A CLOSE AIR SUPPORT AIRCRAFT FOR TOMORROW
Cal Poly-SLO

1.0 INTRODUCTION

The close air support mission has changed little over time. The means of carrying out the mission have, however, changed considerably. The prime objective of the mission, as the term 'close air support' implies, is to deliver air to ground ordinance precisely on target in the presence of friendly forces.[1] The CAS fighter of the past was typically a forward based slower, maneuverable aircraft able to reach the battlefield on brief notice. Fast moving jet aircraft generally required too much scramble time and upon reaching their point of call could not visually identify their targets due to their rate of closure.

While the mission has remained the same the rules of the game have been altered due to the advent of technological advances. New more accurate and reliable weapons delivery systems now allow the pilot of the CAS aircraft to positively acquire the target and deliver ordinance to it with pinpoint accuracy. This advancement significantly reduces the chances of hitting friendly forces and improves air support effectiveness. At the same time, however, advances in anti-aircraft artillery and missiles now pose an even greater risk. In compliance with ground warfare procedure, the majority of CAS missions will very likely take place under the cover of darkness to preserve the advantage of surprise.[1] This requires the aircraft to be able to see and effectively navigate without the benefit of light. In order to survive in the presence of the modern threat the aircraft must remain unseen by the enemy, requiring the capability to operate at low level in unfamiliar territory. The Cyclone close air support aircraft possesses the capability to carry out the mission.

2.0 MISSION DESCRIPTION

The Cyclone aircraft is designed to meet the requirements of three specified military missions. These include a low level, high-low-low-high and a ferry mission.

The low level mission is the primary design mission consisting of five legs (Figure 5.1).

A. Engine warm-up, taxi, take-off and climb-out. Equivalent to five minutes at military power.

B. Dash at sea level to a point 250 nautical miles from take-off point.

C. Combat: two passes at sea level on military power speed less 50 knots.

Combat passes include a 360 degree turn and a 4000 ft. energy increase.

Release air to ground weapons.

D. Dash 250 nautical miles on military power.

E. Land with 20 minutes fuel reserve.

The high-low-low-high mission has the same objective as the low level mission, but includes a best altitude leg for increased fuel efficiency (Figure 5.2).

B. Climb-out at military power to best altitude for cruise.

C. Cruise at best altitude and speed for 150 nautical miles.

D. Decent to sea level.

E. Loiter at best speed at sea level.

F. Dash at sea level on military power for 150 nautical miles.

G. Combat: two passes at sea level on military power speed less 50 Combat passes include a 360 degree turn and a 4000 ft. energy

H. Dash on military power 100 nautical miles.

I. Climb to best altitude for cruise.

J. Cruise at best altitude and speed 150 nautical miles.

K. Descend to sea level.

L. Land with 20 minutes reserve fuel.

The ferry mission is designed for maximum range without use of in-air refueling (Figure 5.3).

A. Engine warm-up, taxi, take-off and climb-out. Equivalent to five minutes at military power.

B. Climb to best altitude and speed.

C. Cruise at best altitude and speed for 1500 nautical miles.

D. Descend to sea level.

E. Land with 20 minutes of fuel reserve.

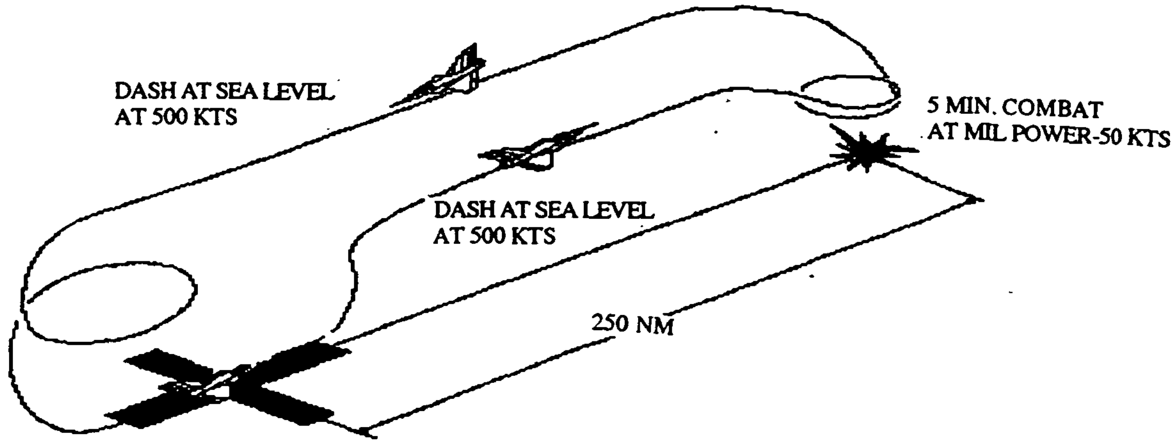


Figure 2.1 CAS Design Mission

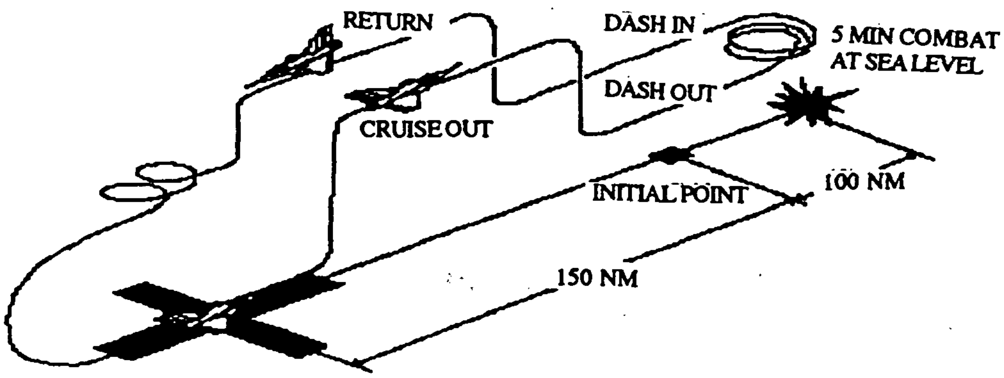


Figure 2.2 High-Low-Low-High CAS Mission

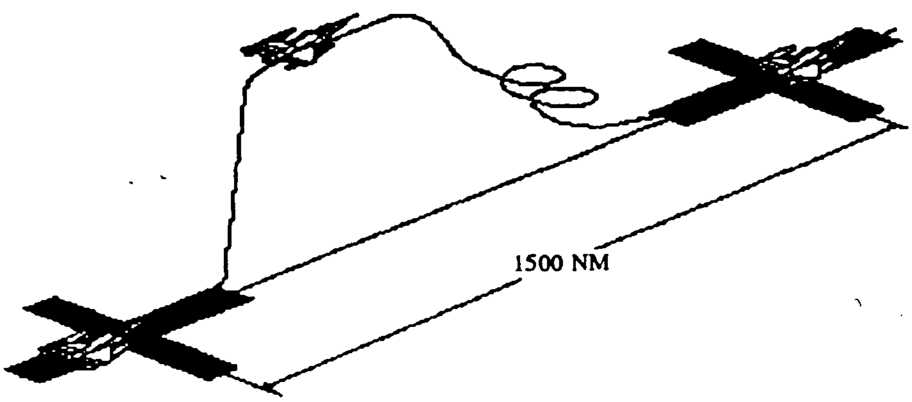


Figure 2..3 Ferry Mission

3.0 PERFORMANCE

3.1 PERFORMANCE OVERVIEW

All performance calculations and analyses of the Cyclone were done for standard day and atmosphere. Analyses was performed using methods in Ref. 5, 6, and 7. The Cyclone geometry, weights, aerodynamics, propulsive data; and mission specifications were used for the analyses. The performance analyses were done to meet the following missions:

1) Low Level Mission
2) High-Low-Low-High Mission
3) Ferry Mission

and additional performance requirements. The Low Level mission is the design mission of the Cyclone.

Cyclone is powered by two low bypass, augmented engines which were scaled to 158% of a rubber engine. The engine data, which is in Appendix A, represents advanced technology mixed flow turbofan engine. Improved, future engines will be considered for the Cyclone.

3.2 MISSION PERFORMANCE

With Low Level mission as the design mission of the Cyclone, the determined design mission take-off and fuel weights were used for all other missions and additional requirements. The gross take-off weight is 54527 lbf with a fuel weight of 12797 lbf and a payload weight of 13552 lbf. The maximum velocity at sea level is limited to 652.8 knots, which is Mach 0.976, due to structural loads and propulsion constraints. The design, maximum cruise speed is 544 knots which is Mach 0.827. All missions require a dash and combat speeds of 500 knots and 450 knots respecitvely. At sea level, the maximum rate-of-climb(ROC) at combat weight is 9600 ft/min. with maximum military power

and 31407 ft/min. with augmented power at the respective flight speed of 361.4 knots and combat speed of 450 knots.

3.2.1 OPTIMUM FLIGHT CONDITION AND ABSOLUTE CEILING

The best altitude and best speed of the Cyclone is 20,000 feet and Mach 0.6 respectively. These values were calculated for best endurance. The corresponding L/D is 10.4 and TSFC is 0.69 lb/hr/lbf. All missions require the Cyclone to cruise at the best altitude and speed, and dash at sea level. The thrust specific fuel consumption at sea level and 20000 feet, which are the critical altitudes, are presented in Figure 3.2.1.

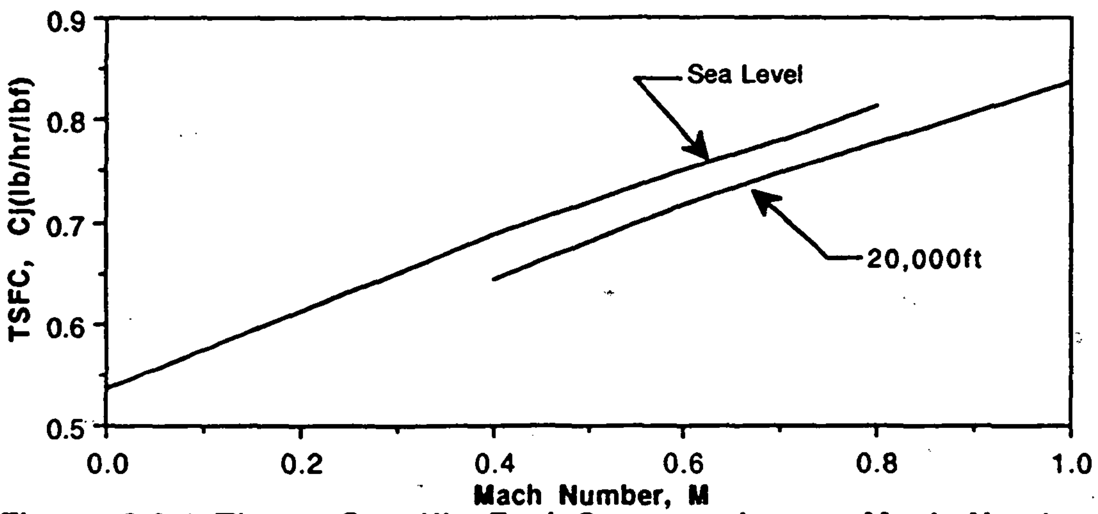


Figure 3.2.1 Thrust Specific Fuel Consumption vs Mach Number At Sea Level and 20,000ft

The absolute ceiling at maximum military power is 28,000 feet as shown on the altitude and maximum rate of climb plot in Figure 3.2.2. The maximum rate of climb was evaluated at the combat weight of 31903 lbf, which includes self-defense stores and 50 percent of the internal fuel. A comparsion of maximum ROC with military power and ROC at Mach 0.8 with augmented power is presented in Figure 3.2.3.

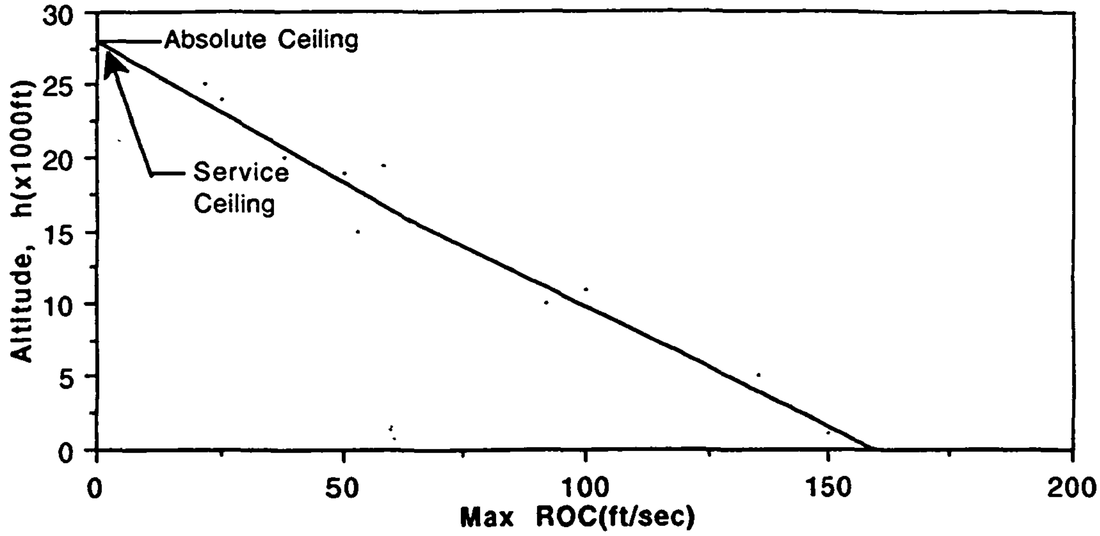


Figure 3.2.2: Absolute and Service Ceilings

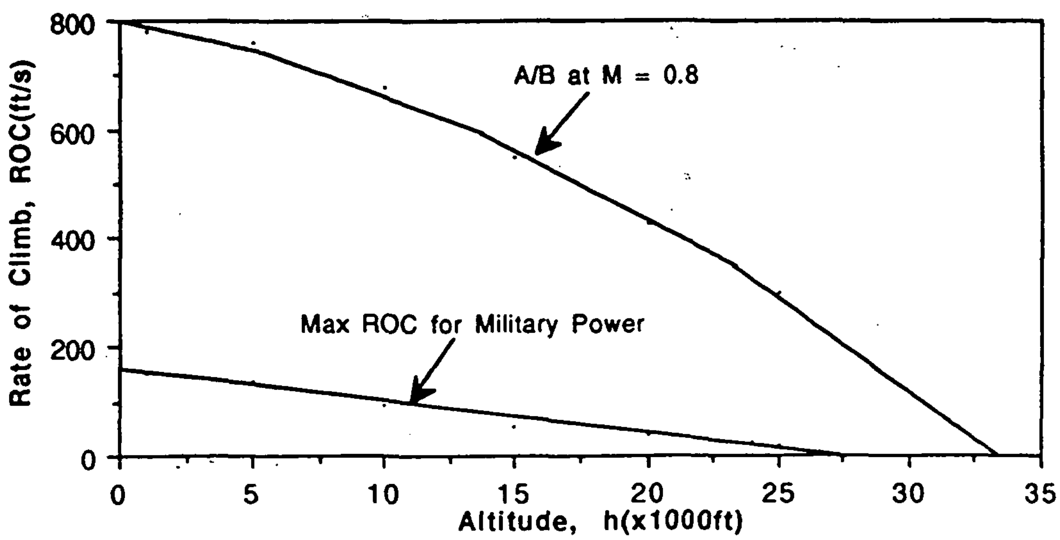


Figure 3.2.3: Rate of Climb at Altitudes

Because the maximum required thrust was met with the maximum augmented or afterburner power, the absolute ceiling is relatively low.

3.2.2 RANGE

Using the 12797 lbf of internal fuel of the Cyclone, the range with various payload weights is shown in Figure 3.2.4. The designed payload weight was 13552 lbf. The range calculation assumed 6 percent of the fuel will be used for take-off and climb, and used a cruise TSFC of 0.69 lb/hr/lbf and L/D of 10.4 at best altitude and speed. The assumption of 6 percent of fuel used was used to make the range conservative. The initial weight included the payload and fuel weight, while the final or landing weight included the payload weight and fuel weight for 20 minutes of loitering.

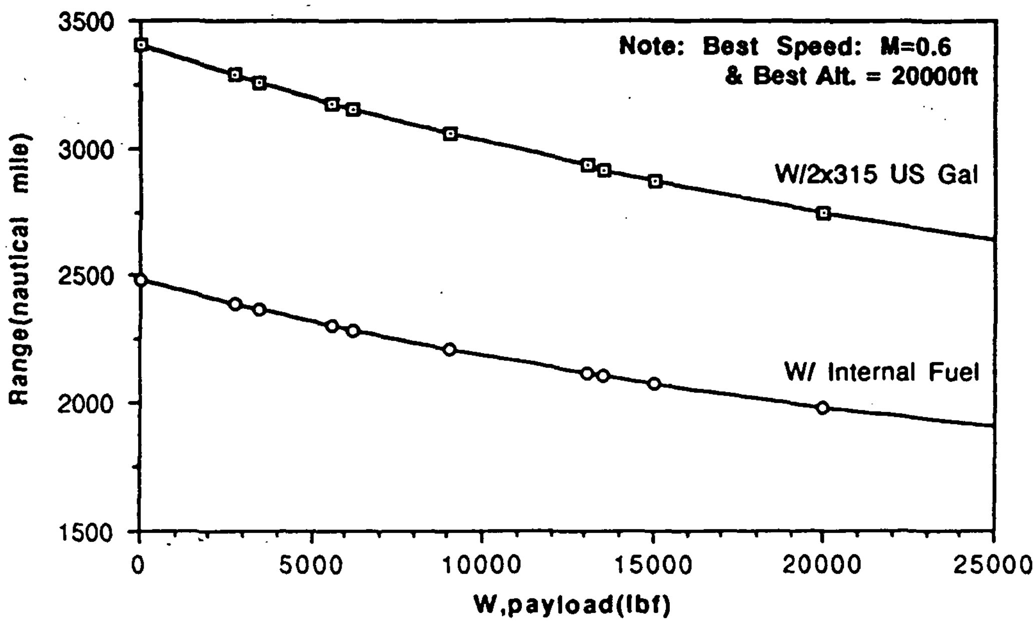


Figure 3.2.4: Cyclone-Range vs Weight of Payload

3.2.3 SUSTAINED AND INSTANTANEOUS LOAD FACTORS & TURN RATES

The mission requirements specify a sustained load factor of 4.5 g's and instantaneous load factor of 6.0 g's at combat speed and sea level. The structural maximum normal load factor is 7.5 g's. The level, pull-up, and pull-down turn rates are at the required load factors and combat speed of 450 knots. The turn rates will increase with greater load factors. The load factors and turn rates are presented in Table 3.2.1.

Table 3.2.1 Load Factors and Turn Rates

1)Load Factor: 4.5 g's	Turn Rate (deg/sec)
Level Turn	10.8
Pull-Up	8.6
Pull-Down	13.5
2)Load Factor: 6.0 g's	
Level Turn	14.5
Pull-Up	12.3
Pull-Down	17.2

3.2.4 FLIGHT ENVELOPES

The specific excess power maps of the Cyclone were determined using the available maximum military and augmented power, and required power at various Mach numbers and altitudes. These maps are presented in Figure 3.2.5, for maximum military power and in Figure 3.2.6 for max augmented power. The specific excess power was evaluated at the combat weight. These two specific excess powers can be compared in Figure 3.2.7, which shows the flight envelope of the Cyclone.

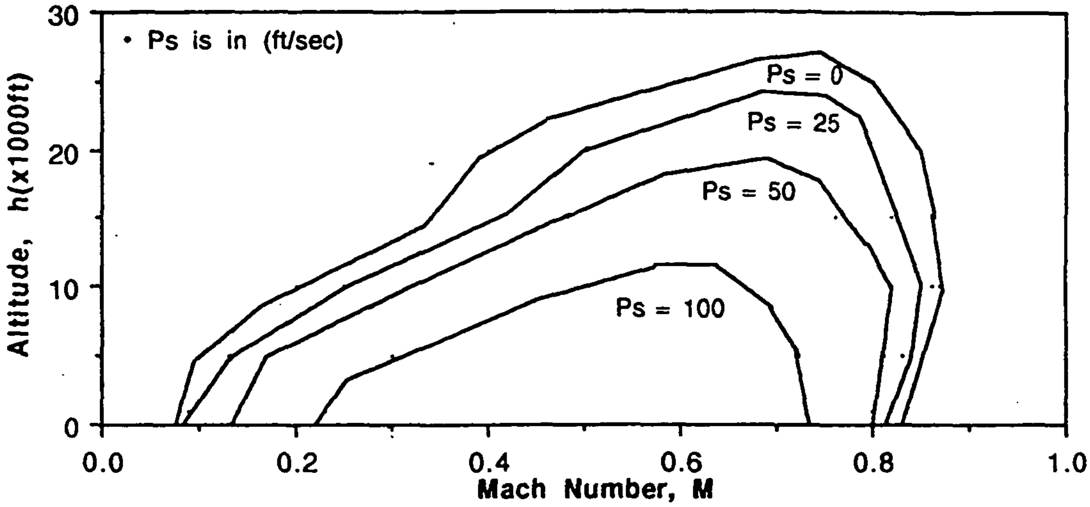


Figure 3.2.5: Specific Excess Power Contour For Maximum Military Power

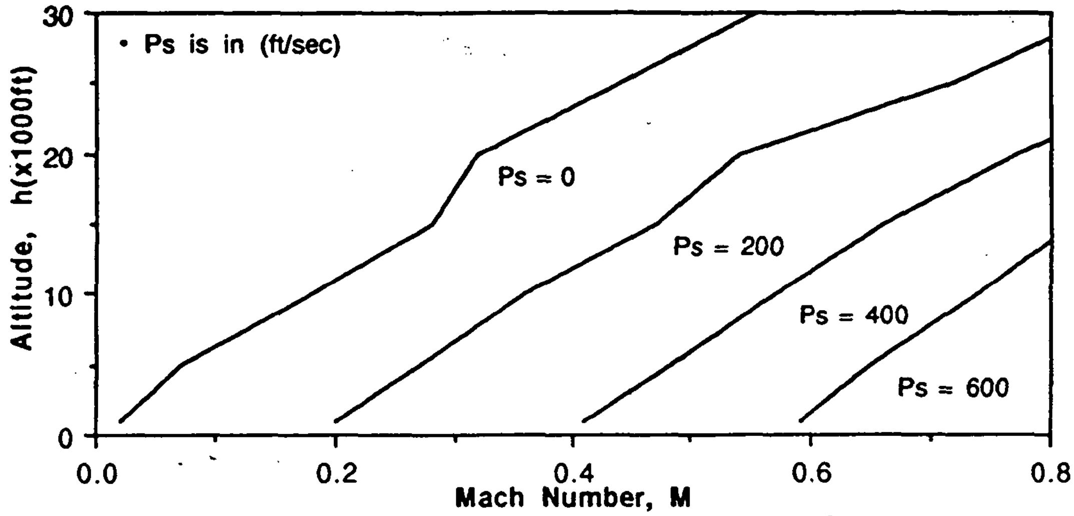


Figure 3.2.6: Specific Excess Power Contour For Maximum After-Burner

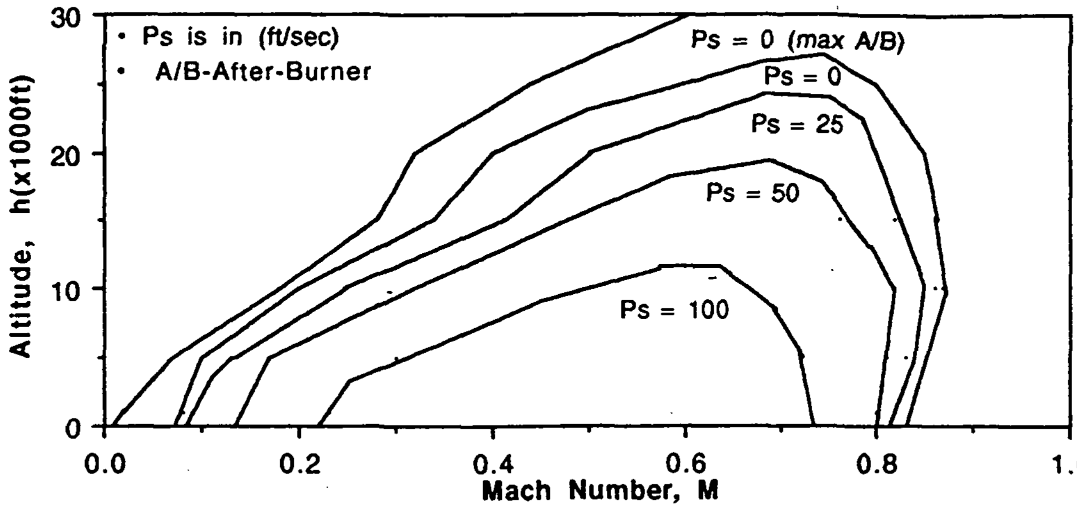


Figure 3.2.7: Specific Excess Power Contour For Max Military Power & Zero A/B Ps

3.2.5 ACCELERATION AND RE-ATTACK TIME

At combat weight, the Cyclone is required to accelerate from Mach 0.3 to 0.5 at sea level in less than 20 seconds and to have a re-attack time of less than 25 seconds. The re-attack time is the time from the first to the second pass weapons release. Each combat pass consists of a 360 degree sustained turn and a 4000 feet energy increase. At Mach 0.3, the drag is 3141.4 lbf and the available thrust is 32545.3 lbf at maximum military power. Cyclone can exceed the required acceleration time at maximum military power. With a ROC of 31407 ft/min. at augmented power and a level turn rate of 27 deg/sec at a speed of 180 ft/sec, the Cyclone re-attack time also exceeds the requirement. Table 3.2.2 summarizes the acceleration and re-attack time performances.

Table 3.2 2 Acceleration and Re-Attack Times

Performance:	Required	Achieved
Acceleration(M=.3 to .5)(sec)	20.0	18.5
Re-Attack time(sec)	25.0	21.0

3.2.6 TAKEOFF, LANDING, AND LOITERING PERFORMANCE

Cyclone is required to takeoff and land with a ground roll distance of less than 2,000 feet on a standard day. Takeoff weight is 54,527 lbf and the landing weight is 52,531 lbf. The landing weight for the Ferry mission is 29,079 lbf. The takeoff and landing L/D's are 6.26 and 9.38 respectively. Takeoff and landing operation is assumed to be on a grass strip, instead of a hard and dry strip, to make the performance more demanding. The respective takeoff and landing friction coefficients are 0.035 and 0.30. For a military aircraft the takeoff and landing velocities were assumed to be 1.1 and 1.3 times the stall velocity of 110.2 knots. The lift and drag effects were evaluated at 70 percent of the liftoff and touchdown velocities. Ref. 7 states that a lift of zero can be assumed during ground roll for landing. The Cyclone landing performance calculation, however, assumed a ground roll CL of 0.2 to make the landing distance conservative.

Loitering performance was calculated at best altitude and speed. The required 20 minutes of loitering at sea level for endurance was evaluated at Mach 0.3 with a TSFC of 0.62 lb/hr/lbf and L/D of 10.1. The initial weight used for the sea level loiter was the landing weight. The high-low-low-high mission will have a loiter time of 73 minutes at sea level for its mission loiter phase. This loiter time is included in the total loiter time. The takeoff, landing, and loitering performances are summarized in Table 3.2.3.

Table 3.2.3: Takeoff and Landing Distances, and Loiter Time

1)Low-Level (Design)	Required	Achieved
Take-Off Distance (ft)	2000	1546
Reserve Fuel (min. at SL)	20	29.7
Landing Distance (ft)	2000	1205
2)HLLH Mission		
Take-Off Distance (ft)	2000	1546
Reserve Fuel (min. at SL)	20	140
Landing Distance (ft)	2000	1205
3)Ferry Mission		
Take-Off Distance (ft)	2000	1546
Reserve Fuel (min. at SL)	20	29.7
Landing Distance (ft)	2000	845

3.3. MISSION PERFORMANCE SUMMARY

3.3.1 LOW LEVEL MISSION

The maximum rate of climb was used in the combat phase. An initial weight estimation was used in takeoff and L/D calculations. The Cyclone fuel weight is higher than the total mission fuel weight which makes it more versatile. Table 3.3.1 summarizes the Low Level mission performance results.

Table 3.3.1 Low-Level Mission Performance

Phase	Mach #	Altitude (ft)	Fuel (lb)	Time (min)	Distance (nmi)	L/D	TSFC (lb/hr/lbf)
Takeoff	0.2	sea level	768	-------	--------	6.3	0.62
Dash	0.76	sea level	5293	30	250	4.0	0.796
Combat	0.68	sea level	286	2	--------	4.4	0.779
Dash	0.76	sea level	5293	30	250	3.2	0.796
Reserve	0.3	sea level	901	20	--------	10.9	0.62
Total	-----	-------	12541	82	500	-----	---------

3.3.2 HIGH-LOW-LOW-HIGH MISSION

Cyclone's performance analysis for this mission makes the same assumption as the Low-Level Mission. The extra loiter time is used for the loitering phase of the mission. The climb is assumed to have no range increase, and the descend has no time, range, or fuel comsumption increase. Table 3.3.2 summarizes the mission's performance results.

Table 3.3.2 High-Low-Low-High Mission Performance

Phase	Mach #	Altitude . (ft)	Fuel (lb)	Time (min)	Distance (nmi)	L/D	TSFC (lb/hr/lbf)
Takeoff	0.2	sea level	768	-------	--------	6.3	0.62
Climb	0.54	sea level	165	2	--------	8.2	0.719
Cruise	0.6	20,000	1499	22.4	150	9.2	0.692
Loiter	0.3	sea level	3280	73	--------	12.0	0.62
Dash	0.76	sea level	2117	12	100	3.7	0.796
Combat	0.68	sea level	286	2	--------	4.2	0.779
Dash	0.76	sea level	2117	12	100	3.5	0.796
Climb	0.54	sea level	165	2	--------	6.7	0.719
Cruise	0.6	20,000	1499	22.4	150	7.7	0.692
Reserve	0.3	sea level	901	20	--------	10.9	0.62
Total	-----	-------	12797	167.8	500	-----	---------

3.3.3 FERRY MISSION

As in the previous missions, maximum rate of climb is used. The takeoff and payload weights are 54527 lbf and 13552 lbf respectively. The payload is replaced with fuel. Cruise is at best altitude and speed. Table 3.3.3 summarizes the Ferry mission performance.

Table 3.3.3 Ferry Mission Performance

Phase	Mach #	Altitude . (ft)	Fuel (lb)	Time (min)	Distance (nmi)	L/D	TSFC (lb/hr/lbf)
Takeoff	0.2	sea level	768	-------	--------	6.3	0.62
Climb	0.54	sea level	165	2	--------	8.2	0.719
Cruise	0.6	20,000	24515	366.5	4645	9.2	0.692
Loiter	0.3	sea level	901	20	--------	12.0	0.62
Total	-----	-------	26349	388.5	4645	-----	---------

4.0 SIZING ANALYSIS

The preliminary design sizing was based on similar aircrafts performance characteristics. It was done to estimated a design point of an acceptable thrust-to-weight and wing loading to achieve the design mission requirements. Takeoff and landing performances are the most demanding flight conditions. The estimated, required takeoff and landing thrust-to-weights and wing loadings defined the design area from which a design point can be selected.

The required thrust-to-weights at wing loadings were done for takeoff, landing, cruise and one engine inoperative. Also combat,climb, and gear up/down approach flight conditions were considered. The landing wing loading and takeoff thrust-to-weight vary with maximum CL. From the determined parameters above, the thrust-to-weight ratio at various wing loadings for the above flight conditions are shown in Figure 4.1. The results show that as maximum takeoff CL increases as the thrust-to-weight ratio decreases. However, the thrust-to-weight ratio increases as the wing loading increases. The take-off and landing parameters are the most critical in the preliminary design phase. Estimating the required takeoff and landing CL's, a design point and a thrust-to-weight and wing loading was chosen for the preliminary. The design point of a thrust-to-weight of.0.59 and a wing loading of 87.

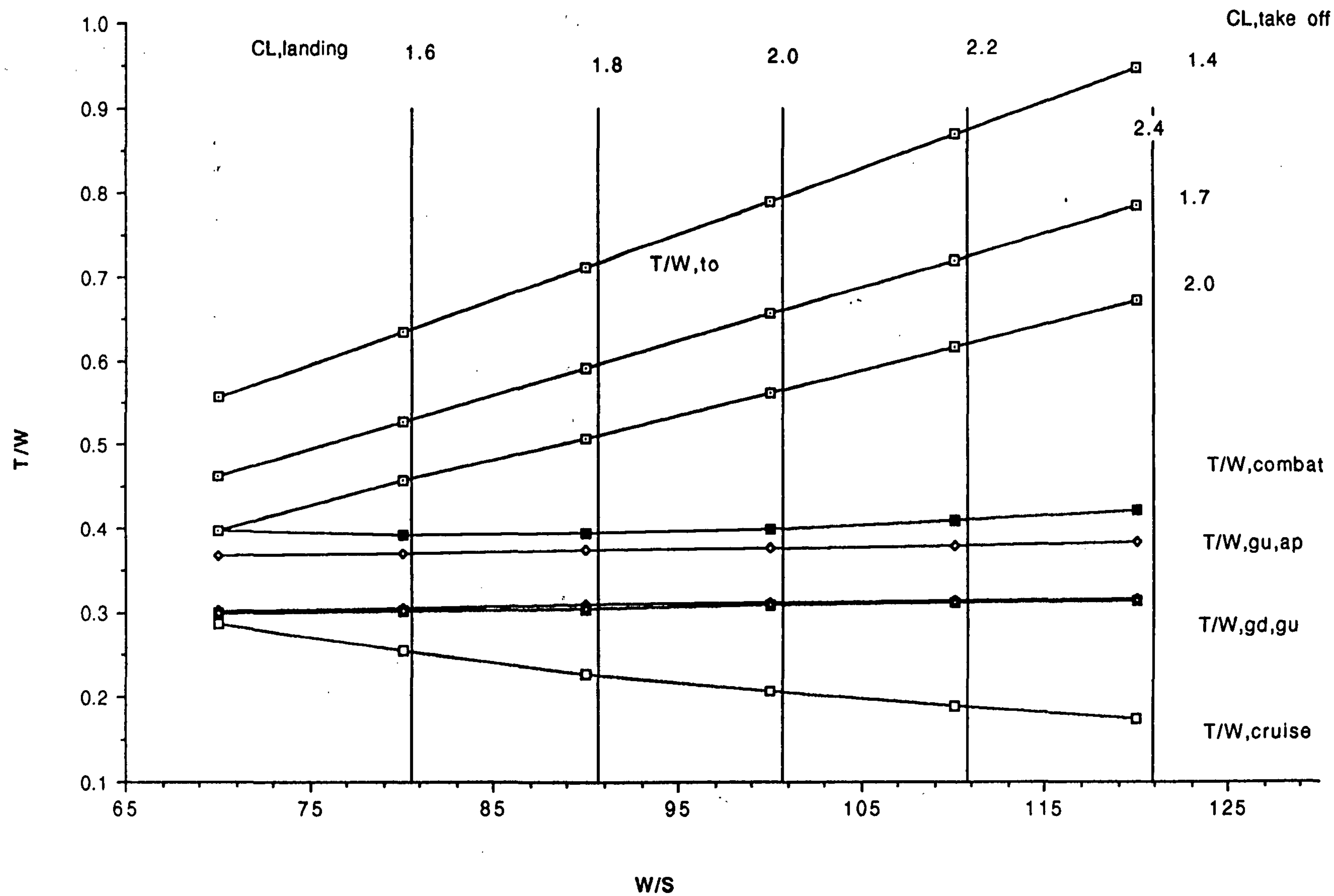


Figure 4.1 Design Area

5.0 CONFIGURATION

5.1 DESIGN TRADE-OFFS

The selection of aircraft configuration was based on the mission requirements as well as fiscal and operational constraints. Various aircraft types were considered including rotorcraft, vertical take-off-and-landing jet, propeller driven fixed-wing and fixed-wing jet..

Rotorcraft, though highly maneuverable and adaptable to small airfields, cannot attain the necessary speeds. Additionally rotorcraft require more maintenance time and spare parts than do fixed wing aircraft. The payload requirement is also typically beyond the capability of all but the largest rotorcraft. For these reasons rotorcraft were rejected as a possible alternative.

Vertical take-off and landing aircraft were also considered. Possessing excellent short strip capability and a high degree of maneuverability, a VTOL could perform the close air support role, however, complexity and comparatively low survivability caused this type to be rejected.

Fixed-wing propeller driven aircraft meet nearly all the requirements with the exception of speed. Close air support aircraft in the past have been of this type, functioning well in the role, however none have carried the weight required of the Cyclone. The required thrust to weight ratio would be difficult to achieve with a fixed-wing propeller configuration.

The fixed-wing jet type aircraft was selected for its ability to meet all the requirements of the mission. Short field capability is relatively poor, yet its load carrying capability and survivability are good. The speed requirement was however the driving factor in the decision to use the fixed-wing jet configuration.

5.2 CONFIGURATION DESCRIPTION

The Cyclone employs is a conventional fixed-wing, blended wing-fuselage configuration. This allows fuel to be carried internally while stores are carried on hard points located on the wing and fuselage. The blended wing also allows sharp corners to be reduced, shrinking the radar cross section. The empennage consists of a 'V' tail, combining the horizontal and vertical surfaces. This reduces surface area and consequently skin friction drag. The propulsion system consists of two turbofan engines in a buried, rear fuselage placement. The engines are spaced apart to enhance survivability in the event that one is hit. The aircraft will be piloted by a single crewman, housed by a bubble canopy and surrounded by cockpit armor.

6.0 COMPONENT DESIGN

6.1 FUSELAGE

6.1.1 FUSELAGE CONFIGERATION

The accommodation of the cannon presented a significant challenge. Much of the inboard profile was dictated by the placement of this gun. The cannon produces considerable recoil, therefore the gun was placed so that the line of action would pass through the center of gravity of the aircraft. Another important design consideration in the design of the fuselage is for pilot visibility and survivability. The canopy cockpit provides the pilot with a nearly panoramic view see Figure 6.1.1.1 A titanium tub was also deemed necessary on the Cyclone even though it adds nearly 1000 lbs to the design. The Cyclone design group also decided to blend the fuselage into the wing, decreasing the interference drag and maximizing the fuselage internal volume. Because of this widening all of the mission fuel requirements are able to be stored within the fuselage. This is most desired in an attack aircraft since the localization of the fuel in the fuselage reduces the target area compared to the more susceptable areas of the wing. Thereby, increasing survivability.

6.1.2 FUSELAGE FINENESS RATIO

The over-all length of the fuselage was dictated by the takeoff rotation that limited the length of the fuselage behind the wing and the gun placement which dictated the length of the fuselage in front of the wing. The fineness ratio of 7.25 was determined by Figure 6.1.2.1. The fuselage drag is minimized with this fineness ratio.[3]

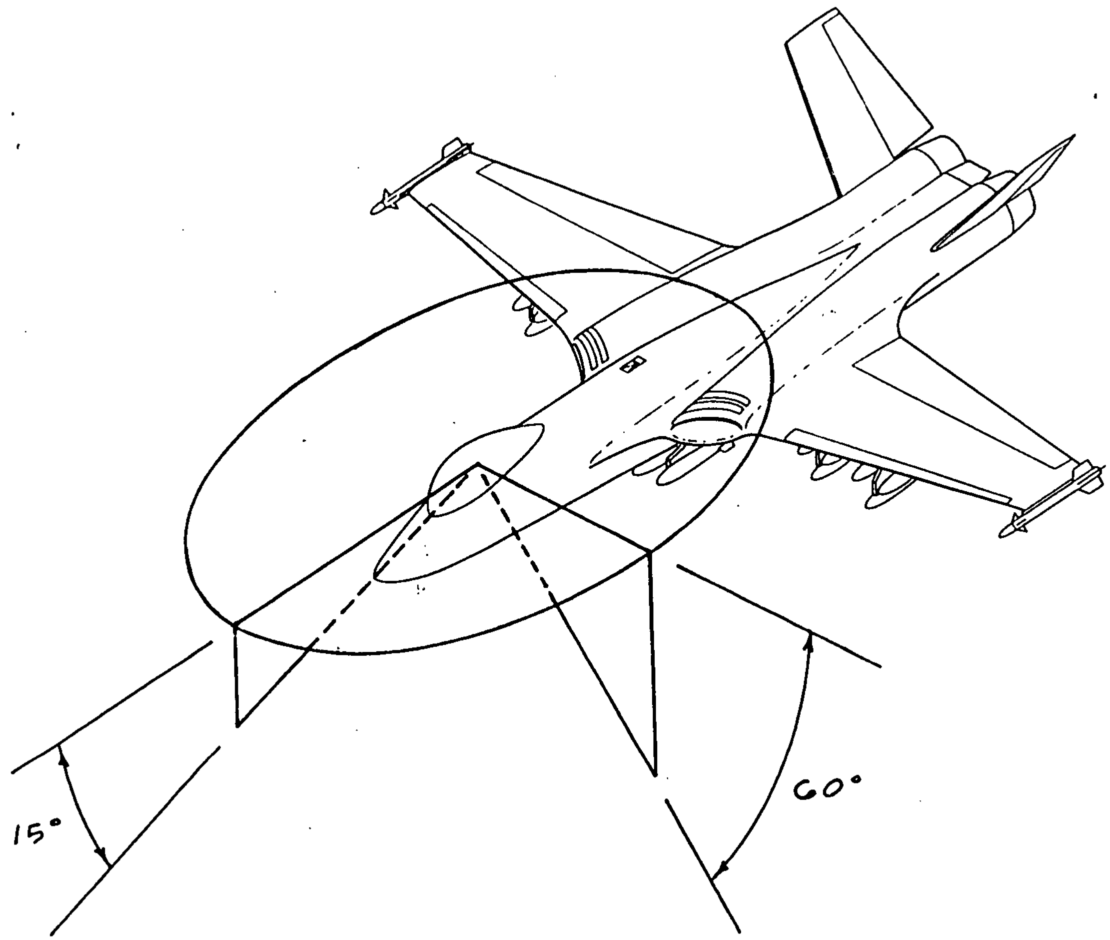


FIGURE 6.1.1.1 PILOT VISIBILITY

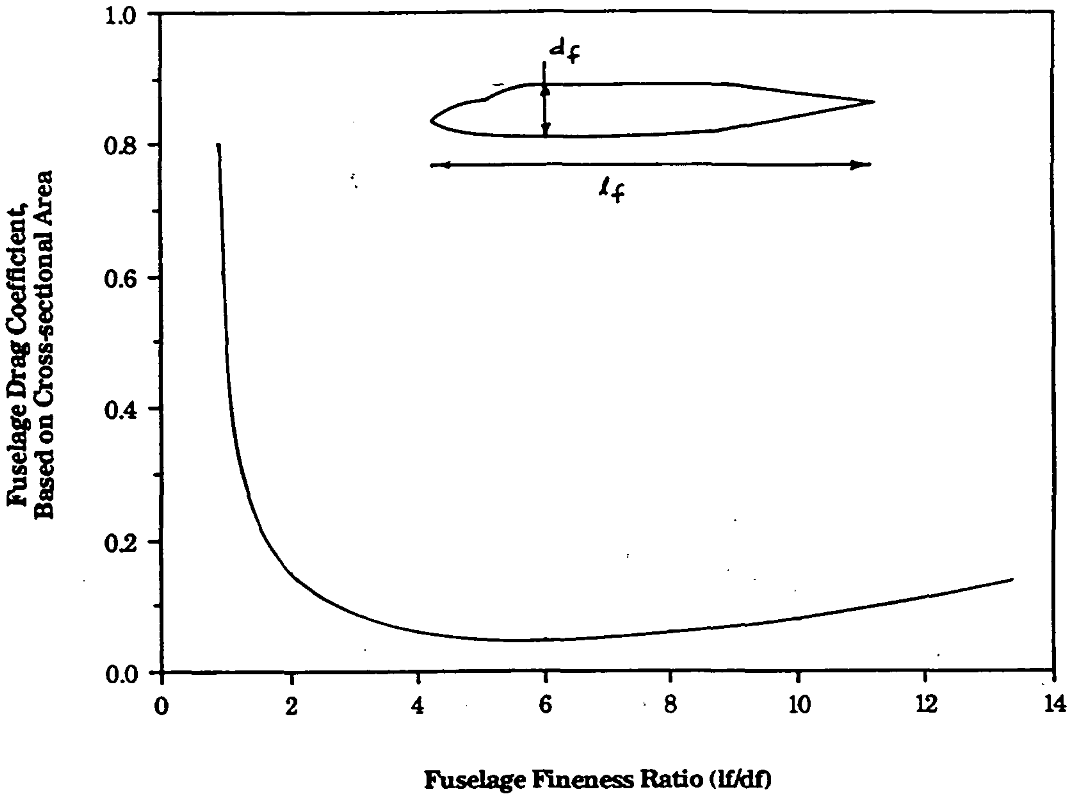


Figure 6.1.2.1 Fuselage Fineness Ratio

(Reference 3)

6.1.3 COCKPIT DESIGN

The Cyclone cockpit is designed with the intent of optimizing the integration of the pilot and aircraft. In order to accomplish this, the pilot must be physically and mentally matched to the machine. He must have the widest range of visibility possible in both azimuth and horizon. Since the Cyclone is designed for air-to-ground operations, the pilot must also have a good look-down view in straight and level flight. The Cyclone cockpit takes these factors into account, placing the pilot high on the fuselage with 360 degrees of visibility in azimuth and 195 degrees in horizon. Mirrors will line the inside of the canopy, allowing the pilot to visually scan the rear of the aircraft without turning his head.

Efficient physical integration of the pilot with the aircraft requires that he be comfortable for long endurance flights and be able to reach all switches even under high-g loading conditions. To reduce pilot workload the hands-on-throttle-and-stick (HOTAS) concept is employed, allowing him to operate the aircraft with minimal body movement. HUD modes and CRT modes may be manipulated using switches on the throttle and center stick controllers.

Management of flight data is a difficult task for the attack pilot. Therefore, a simple display is required to transmit a large amount of data in an understandable manner from the aircraft to the pilot. The head-up-display relays key information to the pilot without requiring him to look down inside the cockpit. Cathode ray tubes display stores and navigation information, reducing the number of gages and dials (See Figure 11.1.1).

Pilot safety is of key importance in the Cyclone design. The escape system consists of a Martin-Baker Mk 10L 'zero-zero' ejection seat, allowing ejection from a zero altitude and zero airspeed condition. The back will be

declined 15 degrees from the vertical. Titanium armor, capable of stopping 20 mm ground fire surrounds the pilot and the cockpit controls. The cockpit environment is air conditioned and pressurized above 10000 feet. Supplemental oxygen is also used.

6.2 WING/HIGH LIFT DEVICES

6.2.1 WING CONFIGERATION

The aspect ratio, taper ratio, thickness ratio and sweep angle. were chosen by trade-off analysis.[3] The primary Motivation for the planform design chosen was that of low cost, low weight and the highest possible wing loading to make the low altitude ride in turbulence more comfortable.

6.2.2 WING PLANFORM PARAMETERS

The design consideration was for the lowest possible weight and cost. From the graph in Figure 6.2.2.1 it can be seen that the smallest sweep angle and the largest thickness ratio yields the lightest weight. Since a supercritical airfoil is used a small sweep angle and a relatively thick airfoil can be used without a large Mach drag increase. With these parameters in mind the existing planform parameters in Table 6.2.2.1 were chosen.[8]

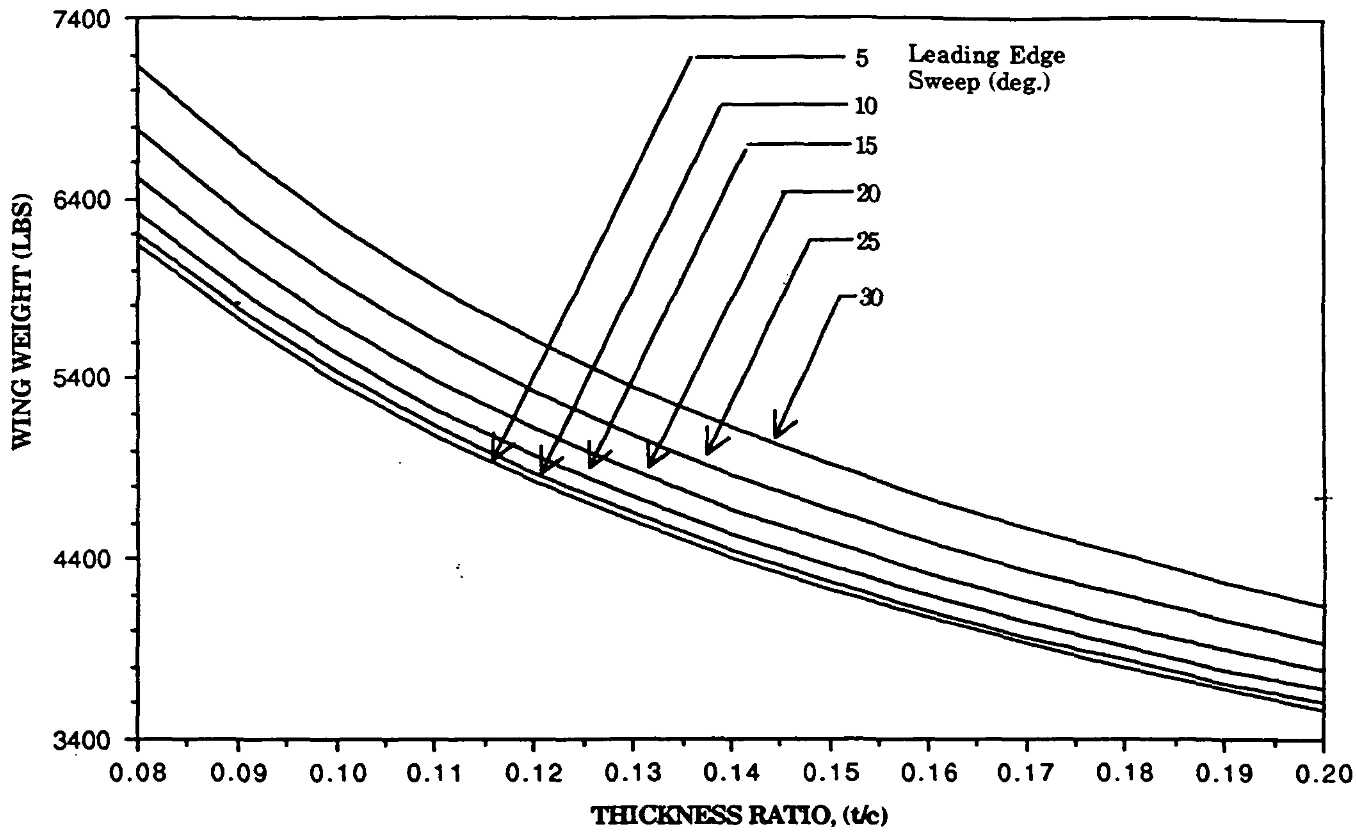


Figure 6.2.2.1, Wing Weight versus Thickness Ratio

Table 6.2.2.1 Main Wing Geometry

Aspect Ratio	5.0
Span	59.3 ft
Surface Area	703.5 ft^2
Sweep Angle	8.0 degrees
Taper Ratio	0.35
Root Chord	16.8 ft
Tip Chord	5.9 ft
MAC	12.9 ft

6.2.3 AIRFOIL SELECTION

Although the maximum speed of the aircraft is subsonic, drag divergence due to Mach number was a major influencing factor in the selection of an airfoil. Originally, the NACA 23012 airfoil section was chosen. The critical Mach number of this airfoil was determined to be only 0.68, however. The next candidate airfoil had a more favorable critical Mach number of 0.775. This matched up well with the maximum Mach number of the aircraft which is 0.756. This airfoil section was designed on the " Panda " software program from a supercritical airfoil in the program's airfoil catalog. The values for lift coefficient given by the software for this airfoil were unreasonably high, though, probably due to the assumptions made by the algorithm for solving the flow solutions. These assumptions included irrotational, inviscid, linearized flow which is unrealistic. Furthermore, no actual data such as that obtained in a wind tunnel was available. So, this airfoil was decided to be an unfavorable choice.

The airfoil section chosen was the CAST 10-2/DOA 2 seen in Figure 6.2.3.1. This airfoil resulted from the Advanced Technology Airfoil Test program which NASA conducted in conjunction with U.S. manufacturers in the early 1980's[9]. It is a 12% thick airfoil with maximum thickness at 40% of the chord. The maximum lift coefficient of this airfoil is 1.1 at an angle of attack of 8 degrees. This can be seen in Figure 6.2.3.2. The critical Mach number for this airfoil before appreciable wave drag occurs is 0.764.

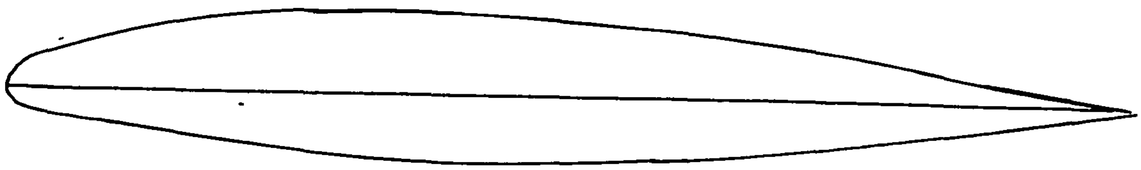


Figure 6.2.3.1-CAST 10-2/DOA Airfoil Section

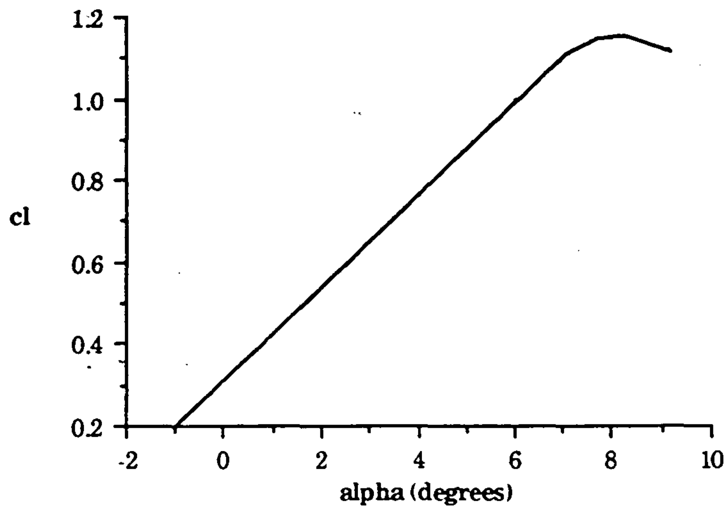


Figure 6.2.3.2-Section Lift Coefficient vs. Alpha

6.2.3.1 FLAP DISPOSITION

Due to less complicated mechanisms required for their deployment, single slotted flaps were chosen as the major high lift device for the aircraft. Although Fowler flaps would undoubtedly have provided a larger increase in lift, the mechanisms associated with their deployment were deemed to bulky, complicated, and unreliable under combat conditions. In addition to the single slotted flaps at the trailing edge, nose flaps will be employed at the leading edge to increase the change in lift coefficient. In combination, these two high lift devices were estimated to provide the necessary increase in lift for the aircraft.

In the sizing of the aircraft, the lift coefficient required for take-off was determined to be 1.7, and the lift coefficient required for landing was determined to be 2.4. With a maximum deflection of 35 degrees and chord fraction of 0.30 for the single slotted flaps, and a maximum deflection of 20 degrees and chord fraction of 0.12 for the leading edge flaps, an increase in lift coefficient of 0.601 was estimated for the aircraft. At the estimated maximum lift condition at an angle of attack of14 degrees, the increase in lift coefficient was estimated to be 1.03, and the maximum lift coefficient for the aircraft was determined to be 1.94 with flaps fully deployed. Since the lift curve slope of the aircraft with the flaps deployed is higher, the increase in lift coefficient with the flaps deployed is greater. These values all make for lift coefficients reasonably close to original sizing values. The flap span for each wing is 13.75 feet with the flap beginning 8 feet from the centerline of the aircraft and going out 21.75 feet from the centerline, as is shown in Figure 6.2.4.1.

6.2.4 CONTROL SURFACE DISPOSITON

For the present Cyclone configuration, the control surfaces, which are the aileron, the elevator and rudder, were sized to provide the required control

powers for aircraft control during maneuvering, trim, and one engine inoperative. The methods of analysis and sizing were done using USAF Stability and Control DATCOM.[10] The flaps and control surfaces are shown in Figure 6.2.5.1. The span of the aileron was calculated to be 75.6 inches, 21% of the aircraft half span, with its inboard location at 260.4 inches from the center-line of the aircraft. Employing a V-tail configuration for the empennage at 38.2 degrees from vertical, the Cyclone has an effective horizontal and vertical span of 256.8 inches and 83.8 inches respectively. The projected elevator span is 65.4 inches with its inboard location at 56.9 inches from the center line, and the rudder span is 83.7 inches of the projected vertical span from the vertical tail root. Using these control surfaces, the control power derivatives were calculated and presented in Table 6.2.4.1. With one engine inoperative, only a constant rudder deflection of 5.7 degrees is needed to maintain aircraft control. These control power derivatives are comparable to existing fighters and attack aircraft.[5]

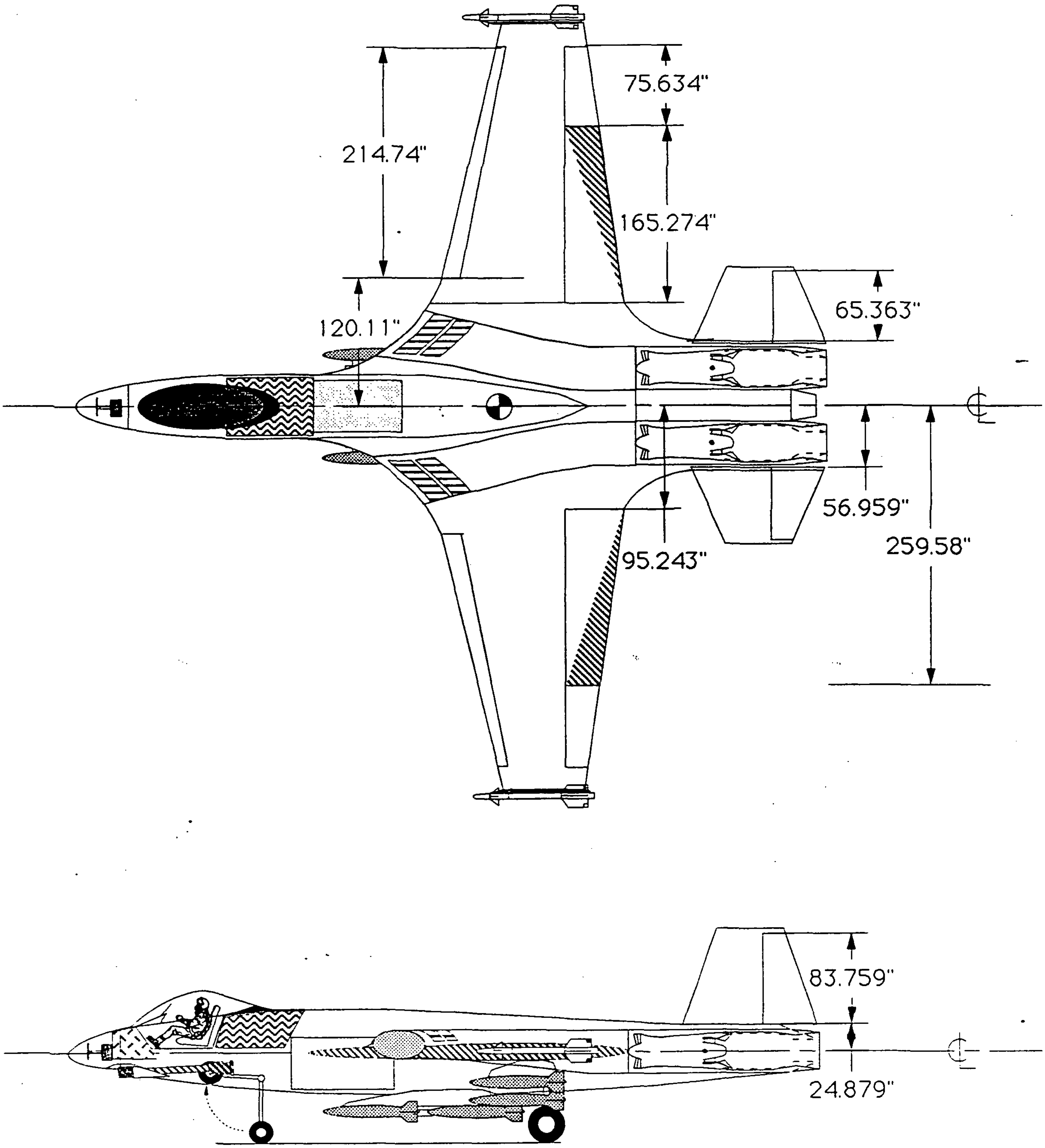


Figure 6.2.4.1 High Lift Devices and Control Surfaces

Table 6.2.4.1 Control Power Derivatives

Flight Condition	Cruise	Combat	Best Speed
Cm ∂e	-0.668	-0.659	-.0638
Cl ∂a	0.800	0.800	0.800
Cn ∂a	-0.00253	-0.00253	-0.00253
Cn ∂r	-0.261	-0.261	-0.261
Cl ∂r	0.280	0.280	0.280
Cy ∂r	0.791	0.791	0.791

6.3 EMPENNAGE

6.3.1 EMPENNAGE CONFIGURATION

Three Empennage configurations were investigated for possible use on the Cyclone. These include a three surface, canard and a conventional design. A canard and three surface design were discarded for the following reasons: due to the anticipated rough field operation the Cyclone group deemed high mounted inlets a priority. With this in mind a canard located in the vicinity of the inlet would create undesirable flow disturbances and degrade the performance of the inlet. The other argument for a canard design is that with two lifting surfaces the cruise L/D can be increased. This argument is invalidated since the Cyclone is a slightly unstable and the tail is designed to lift.[4]

A V-tail arrangement was decided upon due to a small savings in wetted area, thus reducing skin friction drag. Other reasons include a reduction in interference drag and structural weight with only two joining members instead of three or four for conventional designs. The canted outward tails also reduces the profile image of the aircraft, thus reducing the target area and increasing the survivability of the aircraft.

6.3.2 PLANFORM PARAMETERS

The Geometry of the tail surfaces is given in Table 6.3.2.1. The preliminary sizing of the tail was accomplished by comparing similar aircraft tail areas [3]. This method recommends sizing of the tail as if the design had separate horizontal and vertical tails. The butterfly angle was determined by taking the arctangent of the tail areas.

Table 6.3.2 1 Empennage Geometry

Aspect Ratio	2.5
Span	19.04 ft
Surface Area	145 ft^2
Sweep Angle	8.0 degrees
Taper Ratio	0.5
Root Chord	10.2 ft
Tip Chord	5.1 ft
MAC	7.4 ft

6.3.3 AIRFOIL SELECTION

The Empennage airfoil selection was accomplished by researching existing designs. The final selection was made after assuring the tail would not stall before the main wing. A NACA 0012 was chosen for its high Cl and stall angle of attack that is 4 deg. above the stall angle of attack for the main wing.[11]

6.4 PROPULSION INTEGRATION

6.4.1 INLET INTEGRATION

The four considerations for the placement of the inlets are gun gas ingestion, foreign object damage (FOD), landing gear and structural interference. The three types of inlet locations investigated were above wing, in-wing and below wing. Low mounted inlets were dismissed due to their susceptibility to gun gas ingestion and FOD. Above wing inlets were considered to minimize gun gas ingestion and FOD. However, due to structural interference, the inlets needed large bending angles which would present high pressure losses at the compressor face; thus, degradation of inlet performance would occur. In-wing inlets provided the best compromise to the above considerations. The inlets are 6 feet off the ground which is high enough to reduce FOD, and the inlets are placed in the wing away from the fuselage to prevent the ingestion of distorted flow from both the fuselage and circulating gun gas.[12]

6.4.2 POWER PLANT SELECTION

The thrust required for take-off and one engine inoperative (OEI) situations are the most restrictive design parameters . At a preliminary design weight of 59,800 lbs and wing loading of 87 lbf/sq. ft., the thrust to weight ratio was .59. With this thrust to weight ratio, each engine must be able to produce at least 17,650 lbs of thrust.

Different types of power plants were explored for the aircraft's propulsive system. The five candidates were piston, turboprop, propfan, turbojet, and turbofan engines. Ramjets were overlooked due to the flight regime and speed in which they need to operate (at least Mach 3). Piston, propfan and Turboprop engines lack the thrust needed to meet the takeoff and OEI requirements. Turbojet engines were

also dismissed due to the low fuel efficiency at lower altitudes and lower Mach numbers. Turbofan engines were found to be the best option. The fuel consumption of a turbofan engine is between that of a turboprop and turbojet engine. Turbofan engines also can provide enough thrust to satisfy the aircraft's takeoff requirements.

A number of existing turbofan engines were considered. Although information on a rubber developmental engine was provided, it is the desire of this design team to produce a low cost aircraft. Therefore, by using an existing engine, the extra cost of developing one may be deferred. A number of non-augmented turbofan engines were looked at; however, none of the engines considered met the 17,700 lbf thrust requirement. Thus, non-augmented turbofan engines were not employed.

In Table 6.4.1.1, four augmented turbofan engines used in todays fighter aircraft are shown. Each engine shown meets the Cyclone's thrust requirements. Although the Pratt & Whitney engines produce more thrust than the General Electric engines, they also weigh more and are physically longer. Since it is always more desirable to have a smaller and lighter engine in order to reduce the overall weight and length of the aircraft, the Pratt & Whitney engines were not implemented.

Table 6.4.2.1 Augmented Engines

	Pratt & Whitney	General Electric	General Electric	Rubber Engine
	F100-PW-100	F-404-GE-402	F-404 RM-12	58%
Thrust (lbf.)*	23450.0	17700	18000	17713
weight(lbs.)	3033.0	2240	2315	2563.5
fan diameter (in)	46.5	34.8	36.5	34.1
length(in)	191.2	158.8	158.8	155.3

* maximum wet power (static)

The General Electric F404-GE-402 and F404 RM-12 were the best candidates for the aircraft's power plants. However, accurate thrust specific fuel consumption data could not be obtained. This resulted in unknown mission fuel weight requirements. Thus, the feasibility of integrating these engines into the aircraft could not be determined accurately.

Due to this lack of information, rubber engine data provided was used for sizing. The engine was scaled up 58% to closely match the dimensions of the General Electric engines. Thus, installation of the F-404 engines could be made if it can be determined at a later date that the the F-404 engines are suitable for this design.

6.4.3 ENGINE DISPOSITION

The two main concerns when determining the engine placement were protection from small arms fire and easy accessibility for maintenance. Due to the size and thrust produced by the augmented engines, the idea of installing them in pods or nacelles was not considered. The best option was to place the engines inside the fuselage. This would also protect the engines from small arms fire. The engines were placed aft for C.G. purposes and easy accessibility for engine replacement or maintenance. In order to decrease the yawing moment in a one engine out situation, the engines were placed only 2 ft. apart as shown in Figure 6.2.4.1.

6.4.4 INLET DESIGN

High pressure recovery along with uniform flow is desired at the compressor face. High pressure recovery at the compressor face will minimize the compressor's work in compressing the flow to the desired combustion pressure.[12] Uniform flow is desired because distortions in the velocity profile at the compressor

inlet can severely upset the compressor aerodynamics and may lead to failure of the blades due to vibration.[13]

An inlet area of 5.8 sq. ft. was sized. A subsonic straight through internal compression inlet was chosen instead of an external compression inlet. This was done to reduce the external drag cause by spilled flow which is inherent in external compression inlets.[14] Suck in doors will be implemented in flight conditions (takeoff) where the required capture area is greater than the inlet area. The suck in doors are 6.3 sq. ft. each, and they are located near the entrance of the inlet as shown in Figure 6.2.4.1. The flow from the suck in doors is introduced early in the inlet to reduce the chance of flow distortion at the compressor face. A 3.5 ft. section of zero slope leading into the compressor was added to permit the flow to stabilize and, there by, allowing an even velocity profile at the compressor face.[12]

6.4.5 INSTALLED THRUST

Engine performance is affected by the inlet pressure recovery as well as power extractions. A pressure recovery of 99% was obtained for sea level dash conditions. The ratio of installed thrust to un-installed thrust is shown in Figure 6.4.1.1 for maximum power at sea level.

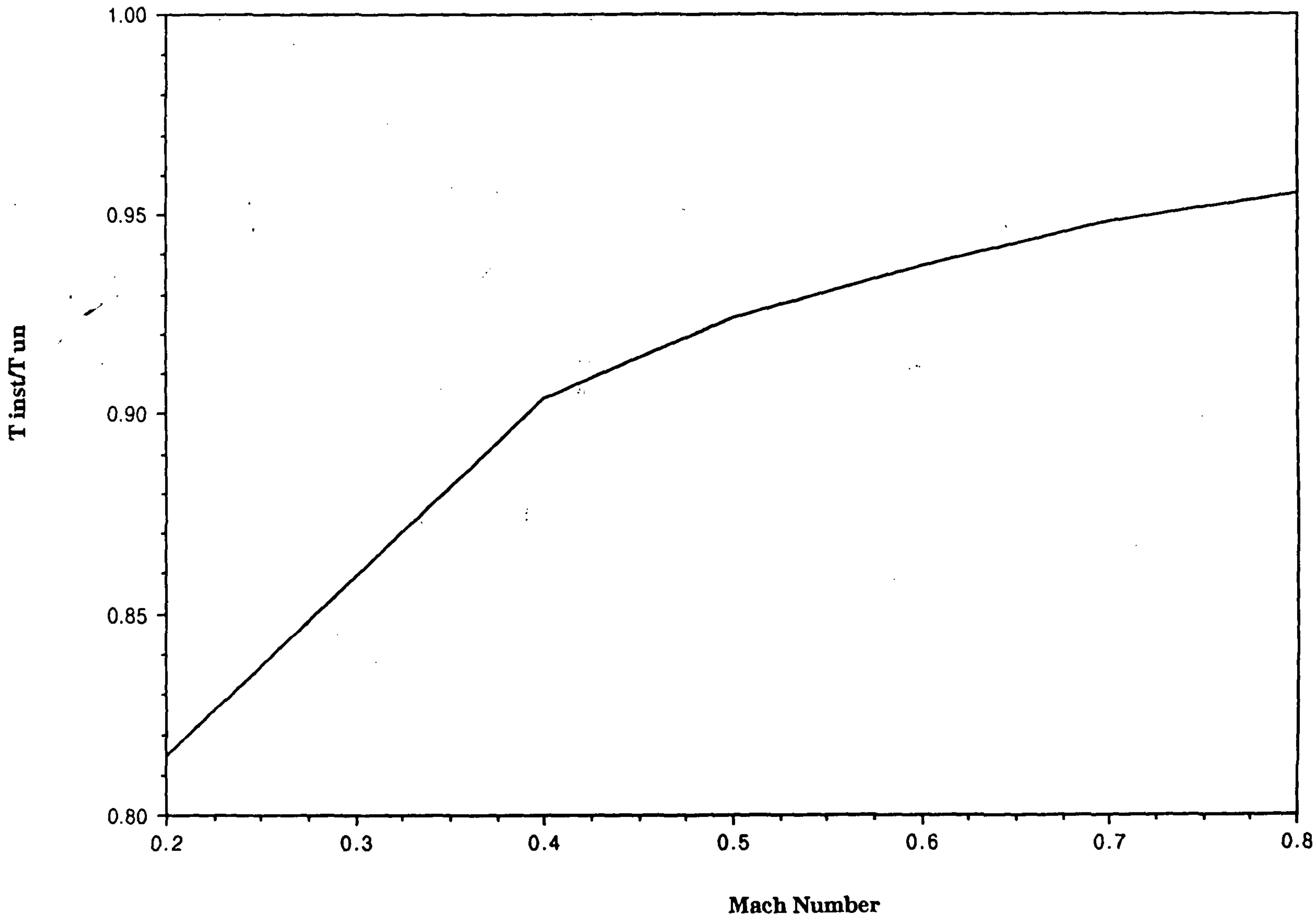


Figure 6.4.2.1 Thrust installation ratio vs. mach at sea level (military power)

6.5 LANDING GEAR

A conventional tricycle type landing gear is installed on the Cyclone because of its good ground maneuvering characteristics and pilot viewing benefits during ground taxi operations. The tricycle gear also provides favorable external weapons load placement capabilities for the Cyclone. Retraction is perfomed by hydraulic actuation. Both the nose gear and the main gear are retracted into the fuselage. The landing gear is designed for a touchdown rate of 17 feet per second[15]

6.5.1 NOSE GEAR

The nose gear location on the aircraft is shown in Figure 6.5.1.1. The nose gear is mounted outboard the fuselage center-line so as to facilitate the cannon recoil to act along the fuselage center-line. The nose gear has one tire and wheel assembly. The nose gear strut houses the liquid spring for landing and ground maneuvering shock absorption. A drag link leading the strut provides the nose gear with additional strength and stability. The wheel and strut are retracted forward and stowed in the cannon bay beneath the aircraft cockpit. Emergency extension is by an integral pressurized pneumatic storage cylinder or gravity drop with the free stream dynamic pressure providing the means for lock down. Table 6.5.1.1 lists the data for the nose gear strut and tire[15].

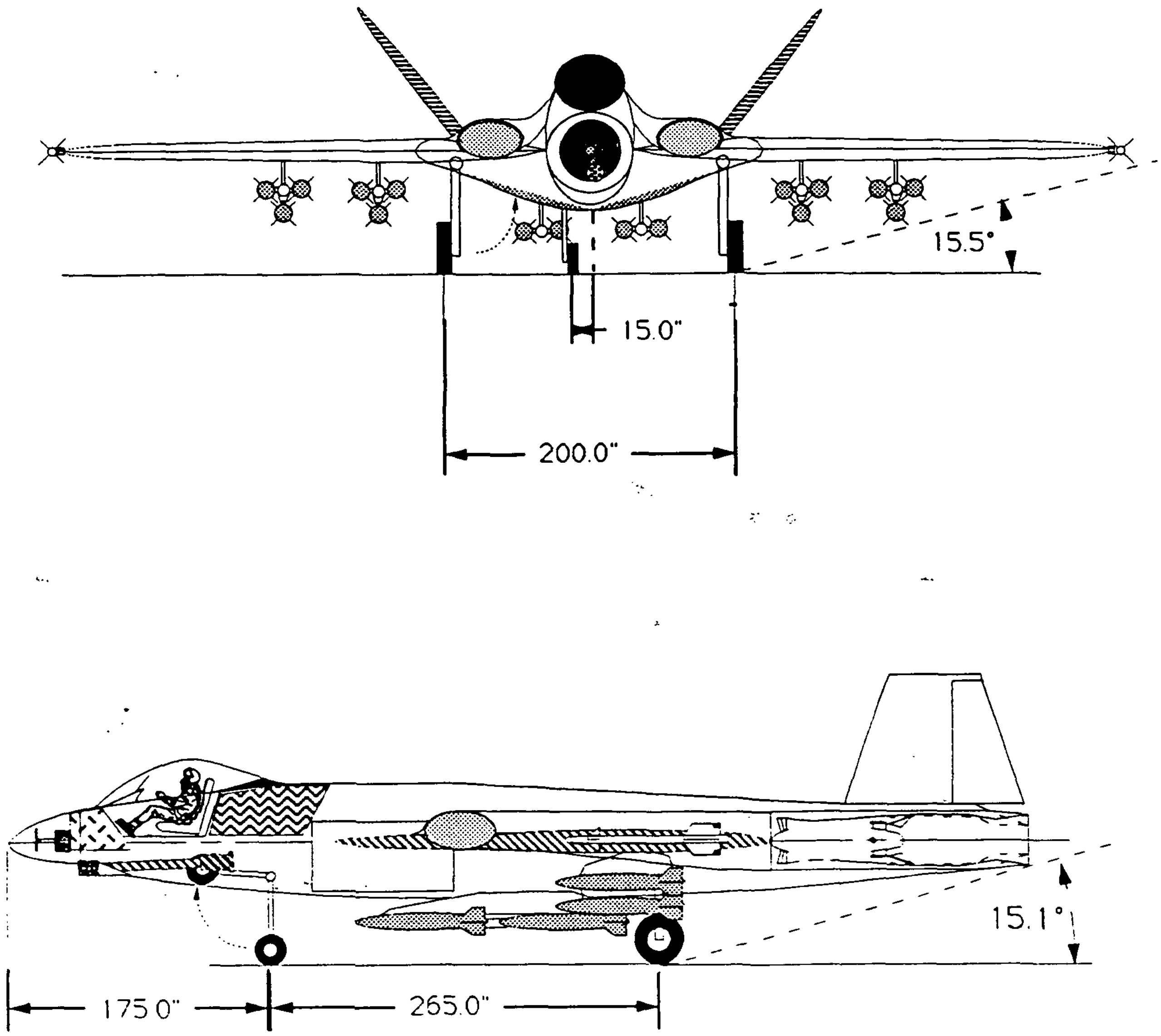


FIGURE 6.5.1.1 LONGITUDINAL AND LATERAL GROUND CLEARANCE CRITERION

Table 6.5.1.1 Nose Gear Data

Strut :		Tire :	
Max static load	5685 lb	Max loading	8700 lb
Max dynamic load	8528 lb	Size	27.75" x 7.65"
Strut length	5.3 ft	Ply rating	12
Strut diameter	0.27 ft	Pressure	130 psi
Strut shock stroke	0.51 ft	Max speed	160 mph

6.5.2 MAIN GEAR

The main gear location is shown in Figure 6.5.1.1. The gear is located near the trailing edge of the wing and is attached inboard of the wing and fuselage junction. The struts and wheels are hydraulically actuated into the fuselage perpendicular to the aircraft center-line . Several advantages are contained within this configuration. The wing structure will weigh less and will be easier and more economical to manufacture because the wheel and strut of the main gear does not cross the wing spars. Additionally, there are no fairings or pods on the wing that would be necessary to house the gear if the main gear were stowed in the main wing. The fairings or pods would produce additional drag. Because the wing is blended into the fuselage, sufficient space is available within the fuselage to stow the main gear.

The 200 inch stance of the main gear improves the aircrafts landing and take-off capabilities in a cross wind and enhances ground taxi stability. Longitudinal ground clearance is a 15.1 degree minimum tail scrape angle. No longitudinal ground clearance problems exist with fuselage or wing bomb loads. Lateral ground clearance is 15.5 degrees with a wing bomb load and 8.2 degrees with a main tire flat and a loss of strut shock absorber hydraulics. Emergency

extension of the main gear is by an integral pressurized pneumatic storage cylinder and gravity assist. Table 6.5.2.1 lists main gear data[15].

Table 6.5.2.1 Main Gear Data

Strut:		Tire:	
Max static load	26708 lb	Max loading	27500 lb
Max KE absorption	244694 lb-ft	Size	39.8" x 14"
Strut length	5.8 ft	Ply rating	24
Strut diameter	0.45 ft	Pressure	145 psi
Strut shock stroke	0.85 ft	Max speed	160 mph

6.5.3 TIP-OVER CRITERIA

The longitudinal tip-over criteria for the Cyclone is shown in Figure 6.5.3.1. The most aft location for the center of gravity (C.G.) has been the main driver in determining the location of the main gear. Space availability for main landing gear placement is significantly reduced aft of the main wing trailing edge which is near the most aft C.G. location. The main landing gear placement results in a 14.8 degree angle between the most aft C.G. location and the main gear as shown in Figure 6.5.3.1.

The lateral tip-over criteria is shown in Figure 6.5.3.2. The lateral tip-over angle for the Cyclone is 42.7 degrees. The wide stance of 200 inches for the main gear gives the aircraft good ground stability on rough fields and minimizes the lateral tip-over angle[3].

6.5.4 RETRACTION SEQUENCE

Nose gear retraction is performed by a hydraulic actuator attached to a moment arm integrally connected at the top of the nose gear strut. Figure 6.5.4.1 shows the retraction sequence as the hydraulic actuator pulls on the

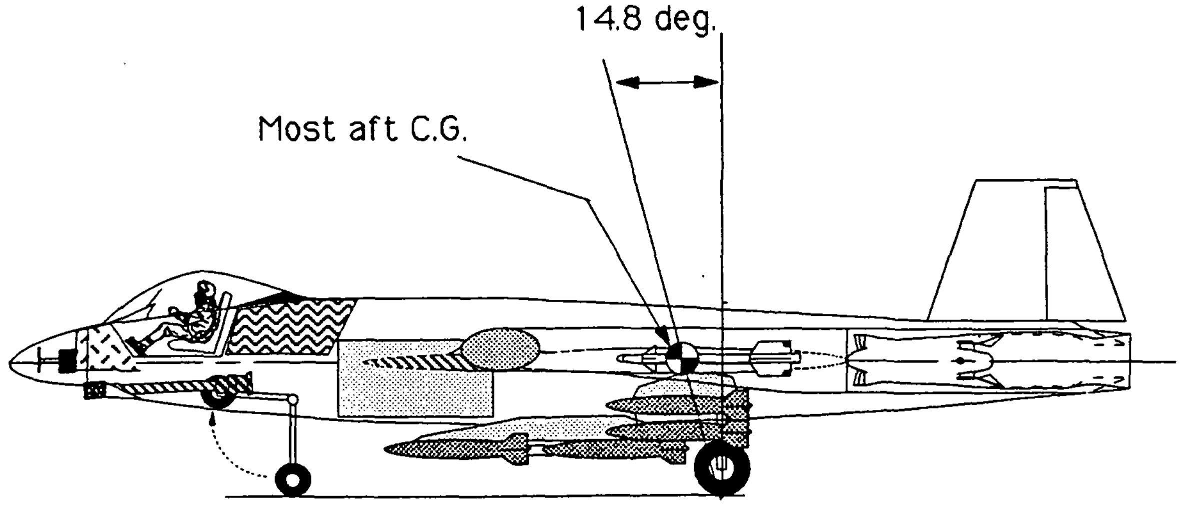


Figure 6.5.3.1 Longitudinal Tip-over Criterion

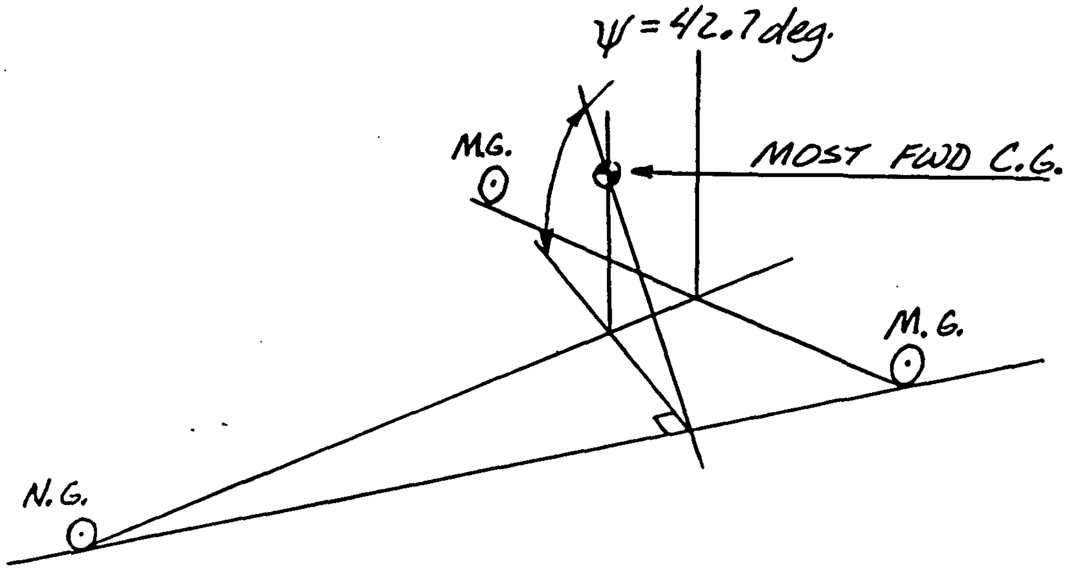


Figure 6.5.3.2 Lateral Tip-over Criterion

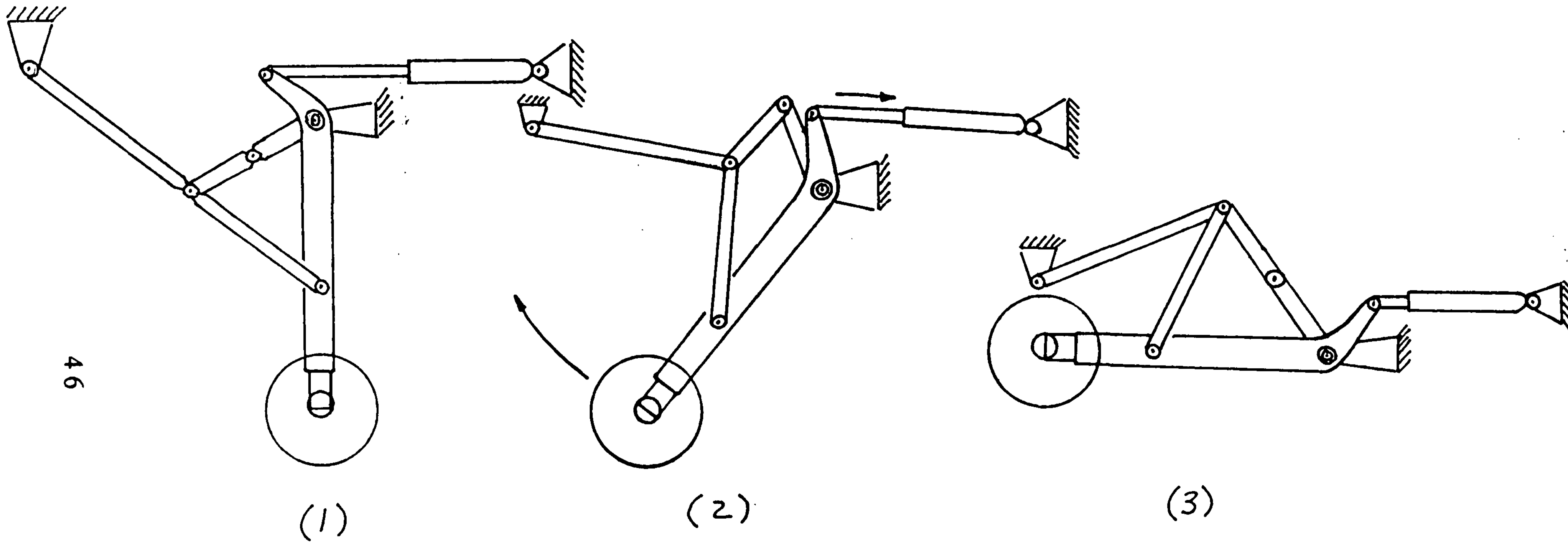


Figure 6.5.4.1 Nose Gear Retraction Sequence

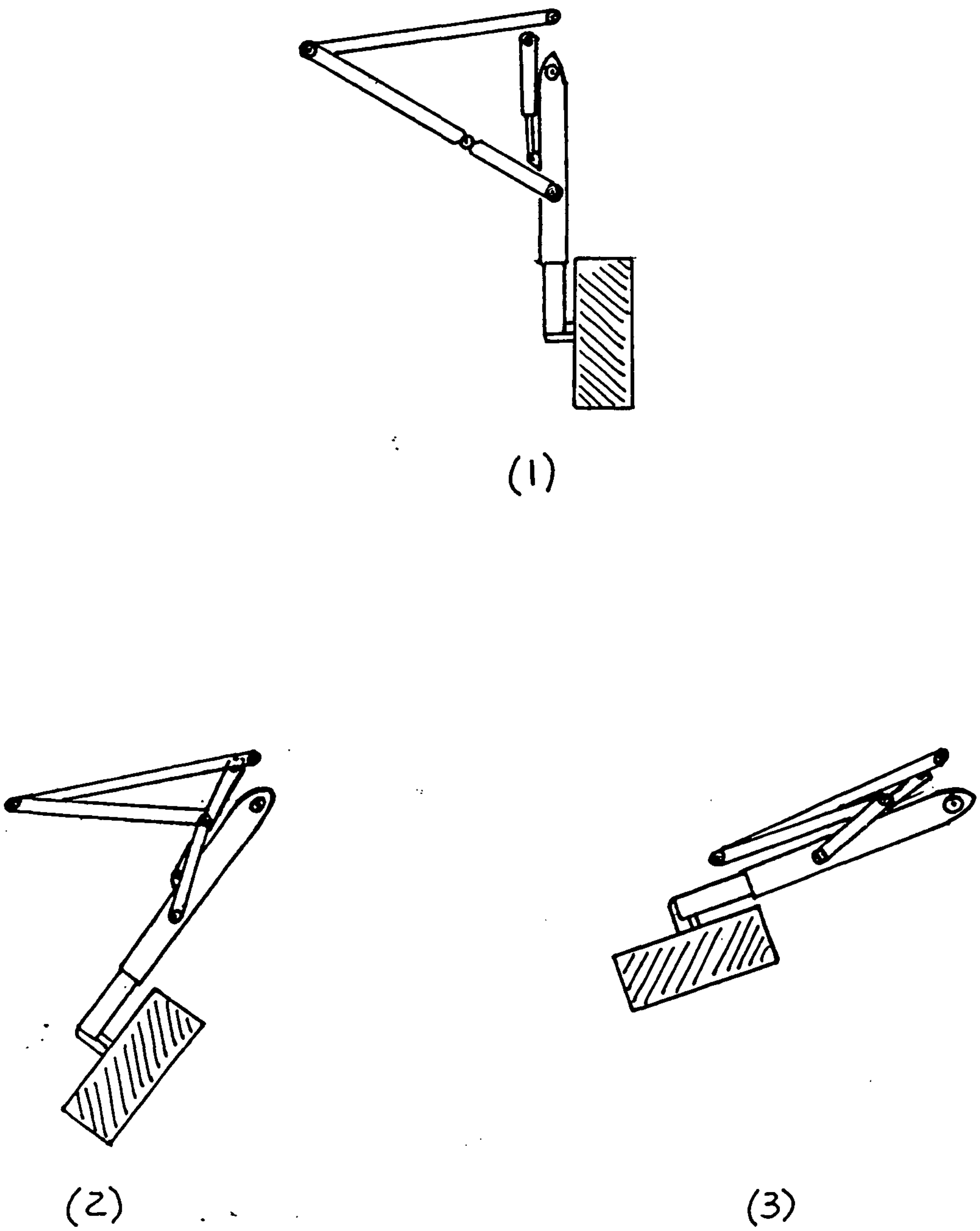


Figure 6.5.4.2 Main Gear Retraction Sequence

moment arm. The strut pivots about a pin joint fixed to the aircraft main structure. The drag link collapses toward the strut with the down lock support link collapsing up toward the nose gear stowage bay. As the wheel and strut rotate upward the drag link and down lock support link continue to collapse and rotate with the strut. The nose gear is stowed slightly above the horizontal requiring the hydraulic actuator to continue pulling on the moment arm and causes the down lock support link to extend.

Figure 6.5.4.2 shows the retraction sequence for the main gear. The main gear strut rotates about a pin joint fixed to the fuselage wing interface structure. The hydraulic actuator is joined on the side and middle of the main strut. When the hydraulic actuator actuates to retract the main gear into the fuselage the side brace collapses out and upward. The retracted stowed angle for the main gear is 21 degrees from the horizontal.

7.0 MATERIAL AND STRUTURES

7.1 MATERIAL SELECTION

A number of structural materials were considered for use in the Cyclone . Composite materials have been gaining acceptance in the aircraft industry do to their 10-20% weight savings over typical aluminium structures. However, the high cost of research, development and manufacturing made this an unacceptable material.[16] Other unacceptable criteria were difficult field repairability deemed necessary for this aircraft. Aluminium Alloys were deemed as the most appropriate material for the Cyclone for their low cost , high strength and ease of manufacturing. Aluminium 7075-T6 is used for the lower wing skin panels and fuselage belly panels due to its superior strength characteristics in tension. This material will also be used for the wing spars, fuselage bulkheads and fuselage longerons due to its rigidity and high strength. Aluminium 2024-T3 will be used in the pressurized cockpit area due to its fatigue resistance. The landing gear strut material chosen is 300M steel because of its superior ultimate stress factor and low cost.[16] Full depth aluminium honeycomb was chosen for the control surfaces and flaps due to its rigidity and low weight. Titanium was deemed necessary as a survivability attribute even though it is expensive and adds considerable weight. This material will be used in a protective tub protecting the pilot from enemy fire, similar to the one used on the A-10 aircraft. Titanium will also be used as protection of vital system components and engine areas. The raydome will be constructed from epoxy fiberglass and the canopy will be a one piece polycarbonate due to its toughness and impact strength.[16]

7.2 STRUCTURAL DESIGN LIMITS

The structure of the Cyclone was designed to the structural limitations set forth in the mission specifications . The critical load limit was found by the following V-n diagrams shown in Figure 7.2.1 The positive load limit is 11.25 g's and the negative limit is -6.75 g's as indicated by the RFP.

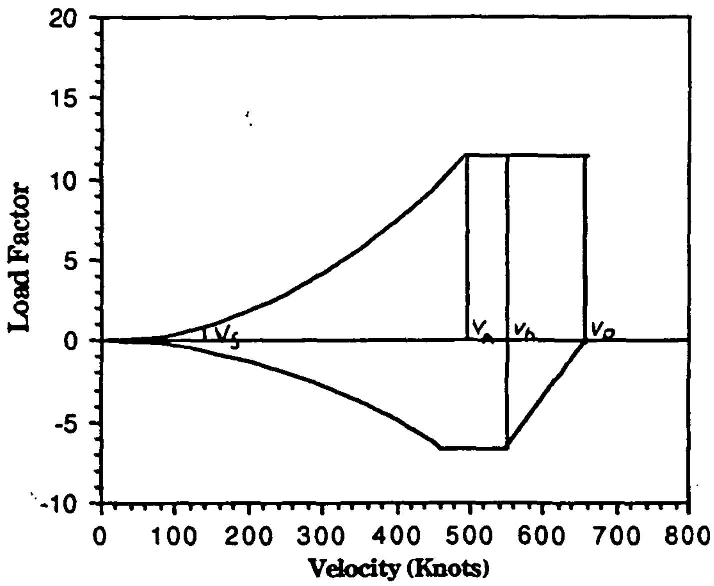


Figure 7.2.1, V-N diagram

The gust load diagram was determined to be unnecessary considering the high maneuver loads for this design.[8] The maximum dynamic pressure is 1000psf which is equivalent to the maximum level flight speed (V_h) of 544 knots at sea-level. For preliminary design it was determined that the positive low angle of attack gave the highest shear and moments on the wing structure. These shear and moments are plotted in Figure 7.2.2 The maximum shear and moment at the root was was determined to be 325,300 lbs and 3,116,016 lbs-ft respectively. Using a maximum tensile stress of 7075 -T6 aluminium of 79,000 psi, and

considering a safety factor of 1.5, a minimum moment of inertia for the wing spars of 9156 in^4 was obtained.

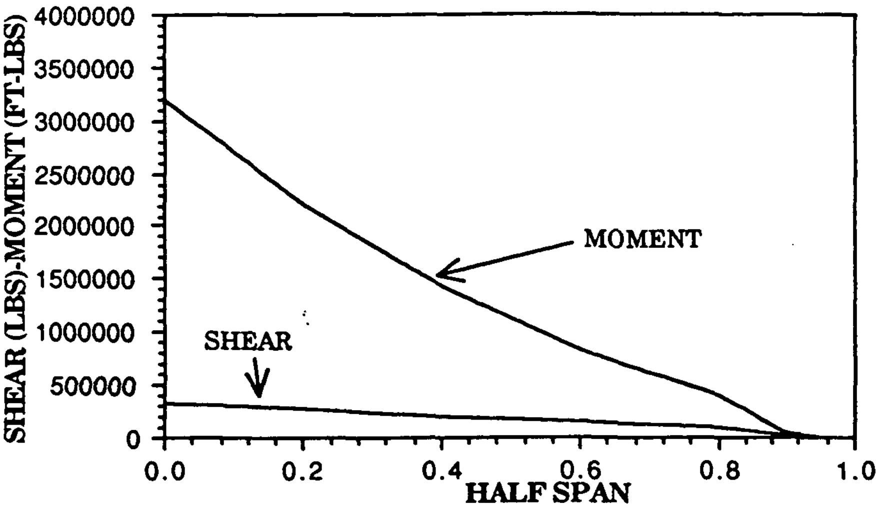
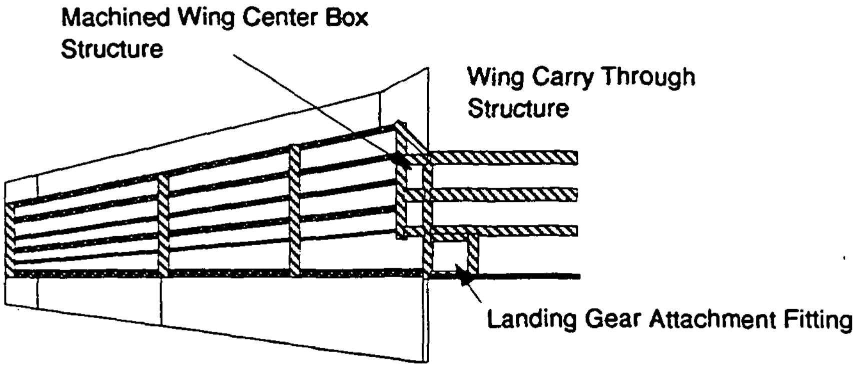


FIGURE 7.2.2, SHEAR-MOMENT DIAGRAM

7.3 STRUCTURAL LAYOUT

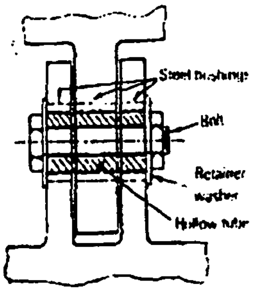
7.3.1 WING STRUCTURE

The preliminary design structural layout of the wing structure is pictured in Figure 7.3.1.1. The wing structure of the cyclone is similar to the McDonnell Douglas F-18 fighter aircraft. The wing structure is a multi-spar design with six spars. This type of structural layout has the advantage of fail- safe design since the aerodynamic loads are distributed more evenly in the six spars.



Wing Structure

Figure 7.3.1.1



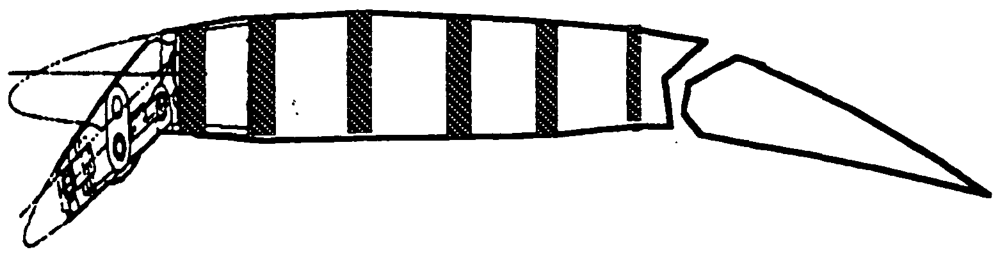
Double Shear Lug with Hollow Tube

Figure 7.3.1.2

7.3.1.2 From Reference 16

The wing spars proposed are integrally machined for their lower manufacturing cost. The preliminary design analysis was accomplished by modeling the spars as "I" beams which yielded a moment of inertia of 1648 in4 each. This inertia was more than sufficient to withstand the critical bending moment determined by the V-n diagrams. These spars connect to a machined wing joining structure pictured in Figure 7.3.1.1 *that transmits the loads to the* airframe via three main carry through structural members. The wing attachment is of the removable double shear lug design with hollow tube for fail-safe considerations. see Figure 7.3.1.2. The wing contains only four main ribs

per half span that transmit the torsion to the main spars. These ribs will also accommodate the bomb rack hard points and the aircraft jack points. The wing skin is machined with tapered thickness along the span to accommodate the increase in bending moment and shear at the wing root. A cross section of the wing structure is shown in Figure 7.3.1.3.



Wing Structure Cross-Section

Figure 7.3.1.3

Not to Scale

7.3.2 EMPENNAGE STRUCTURE

The empennage structural layout is pictured in Fig.. The structural layout is similar to that of the main wing previously discussed. This is also a multi-spar structural layout with three main spars and machined skins. All the previously discussed design reasons also apply here. Structural synergism was obtained by incorporating the structural carry through for the empennage with the engine support structure see Figure 7.3.3.1.

7.3.3 FUSELAGE STRUCTURE

The fuselage structure is pictured in Figure 7.3.3.1 The fuselage longeron and bulkhead placement is based on similar aircraft. The Bulkheads are placed at 20 in spacing and the longerons are at 12 in spacing. Also, a structural thickness of 2" was allowed based on similar aircraft.[4]

Fuselage Structure

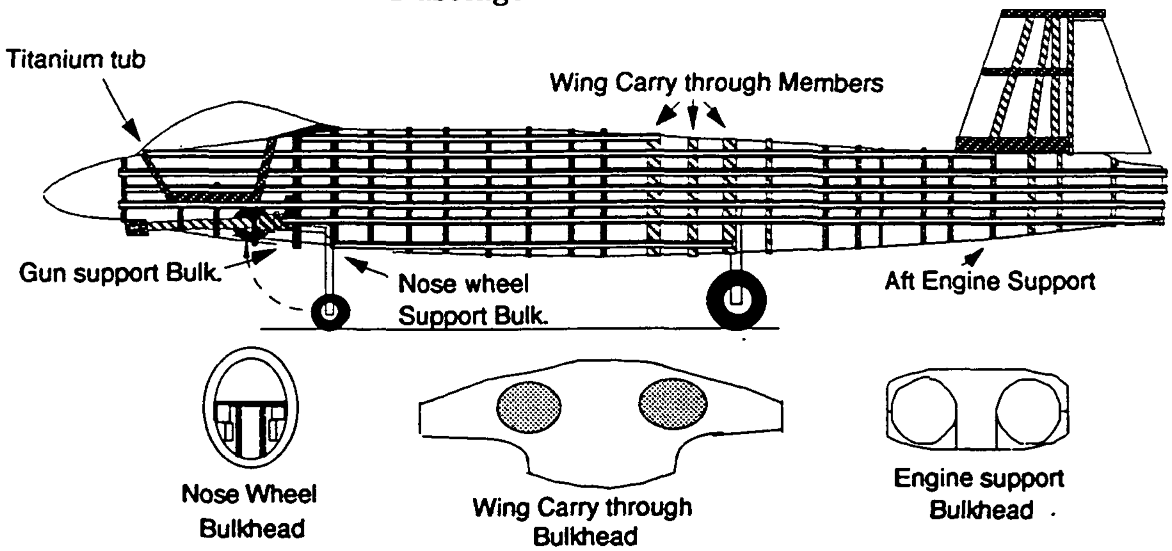


Figure 7.3.3.1

The nose wheel strut is mounted off center to allow for the GAU-8 gun to be placed on the center line of the aircraft. The axial loads caused from the 10,000 lb gun recoil and the landing gear support loads are distributed to the other structure by a rigid channel that runs from the forward radar bulkhead to the forward wing carry- through structure. This structure also supports the gun ammunition canister and gives rigidity to the forward fuselage. Also this center load bearing member will enable access panels in the fuselage sides, where normally, structural members are located. Structural synergism was achieved with the main wing carry through and the main gear strut attachment see Figure 7.3.1.1.

8.0 AIRCRAFT MASS PROPERTIES

8.1 COMPONENT WEIGHTS AND C.G. LOCATIONS

Component weights for the Cyclone were found using empirical methods. The aircraft component weights are based on a maximum take-off weight, and aircraft geometry. The engine weight was scaled from rubber engine data. Maximum take-off weight is 54,527 pounds. Weapons stores, cannon and pilot weights are specified in the RFP. Table 8.1.1 lists the airframe structure component weights and C.G. locations referenced from the aircraft nose.

Table 8.1.1. Airframe Structure Component Weight and C.G. Location

COMPONENT	WEIGHT (lb)	LOCATION (ft)
wing	4558	35
empennage	1256	54
fuselage	4665	26.1
nose gear	360	12.15
main gear	1440	36.81
air induction	850	34.3
titanium tub	1000	10
engine shield	300	49

Propulsion components, aircraft equipment, payload weights, and C.G. locations are in Table 8.1.2, 8.1.3, 8.1.4 respectively.

Table 8.1.2. Propulsion Component Weight and C.G. Location

COMPONENT	WEIGHT (lb)	LOCATION (ft)
engines	5127	48.2
fuel system	573	50
propulsion op. sys	866	50

Table 8.1.3 Aircraft Equipment Weight and C.G. Location

COMPONENTS	WEIGHT (lb)	LOCATION (ft)
refuel sys	44	20
fuel dump	29	50
flight control	1409	50
avionics	690	14.5
hydraulic sys	436	37
oxygen sys	68	45
air & deice	550	24.7
apu	436	45.5
cockpit furniture	202	10
aux equipment	276	35
gun	1840	8.6
paint	164	40
electrical sys	391	40

Table 8.1.4 Payload Weight and C.G. Location

COMPONENT	WEIGHT (lb)	LOCATION (ft)
pilot	200	10
fuel	12797	32
ammunition	2106	21
bomb (wing)x12	6060	35
bomb (fuselage)x8	4040	26.5
missile x 2	400	36.2
missile rack x2	40	36.2
bomb rk(wing)x 4	876	35
bomb rk (fus)x2	438	26.5

8.2 WEIGHT AND BALANCE SUMMARY

Table 8.2.1 lists a summary of aircraft weights and C.G. locations. Figure 8.2.1 shows the Cyclone weight and C.G. excursion from empty condition through typical payload loading and unloading conditions. The in-flight C.G. travel is 14.4% mean chord. The forward C.G. limit is 0.327 mean chord and the aft C.G. limit is 0.486 mean chord. The C.G. travel between the forward and aft limits is 15.9 % mean chord. The forward C.G. limit occurs at the fully loaded condition. The aft C.G. limit occurs when the aircraft is completely empty except for the bomb load on the wings. At take-off with a full load the aircraft is the least unstable. As fuel is used, bombs dropped, and ammunition is discharged, the C.G. travels aft rendering the aircraft further unstable and more manueverable. This is desireable, especially when the aircraft is engaged in combat where inflight manueverability is crucial to successful weapons deployment and aircraft survivability.

Table 8.2.1 Weight and C.G. Summary

LOAD CONDITION	WEIGHT (lb)	C.G.(% chord)
empty	27,526	0.481
bombs (wing)	34,941	0.486
bombs(fuselage)	32,003	0.392
pilot	27,725	0.471
fuel	40,323	0.410
bombs(w) & fuel	47,739	0.422
bombs(w & f) & fuel	52,217	0.371
bombs(w & f) & fuel & ammo & pilot	54,523	0.327
bombs(w & f) & ammo & pilot	41,725	0.350
bombs(w) & ammo & pilot	37,685	0.407
pilot & ammo	29,831	0.395
fully loaded	54,527	0.327

8.3 MOMENTS OF INERTIA

The moments of inertia were calculated assuming that the inertia of each component about its own center of gravity is negligible. The Cyclone moments of inertia for the fully loaded condition are listed in Table 8.3.1. The aircraft is not symmetrically loaded as shown by the nonzero Ixy and Iyz terms.

Table 8.3.1 Cyclone Moments of Inertia

Ixx	67509 slug-ft^2
Iyy	185873 slug-ft^2
Izz	244087 slug-ft^2
Ixy	530 slug-ft^2
Iyz	-12 slug-ft^2
Izx	-9082 slug-ft^2

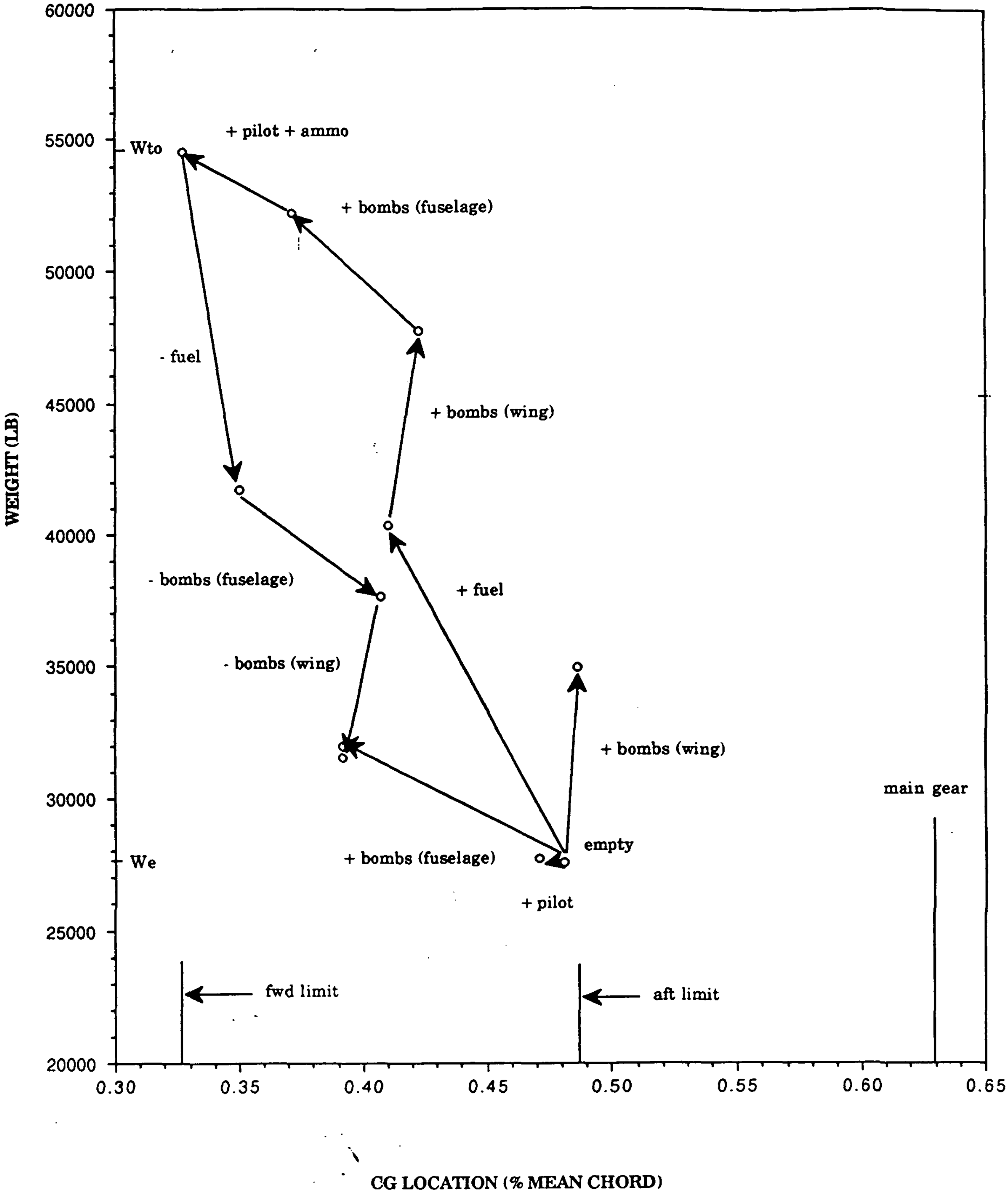


FIGURE 8.2.1 CYCLONE: WEIGHT AND C.G. EXCURSION

9.0 AERODYNAMICS

9.1 LIFT PREDICTION

Figure 9.1.1 shows the relationship between lift coefficient and angle of attack for the aircraft. The maximum lift coefficient obtainable by the aircraft is estimated to be 0.91 at an angle of attack of 18.43 degrees. With the high lift devices deployed, the lift coefficient can be increased by 1.03 at the maximum lift condition. The zero lift angle of attack was calculated to be -4.5 degrees, and the lift coefficient at zero angle of attack was calculated to to be 0.18 for the aircraft[17]. At zero angle of attack, the lift coefficient of the aircraft is increased by 0.601 with fully deployed flaps for a maximum lift coefficient of 0.781. The lift curve slope of the aircraft is estimated to be 2.27/radian without flap deployment and 4.72/radian with flap deployment.

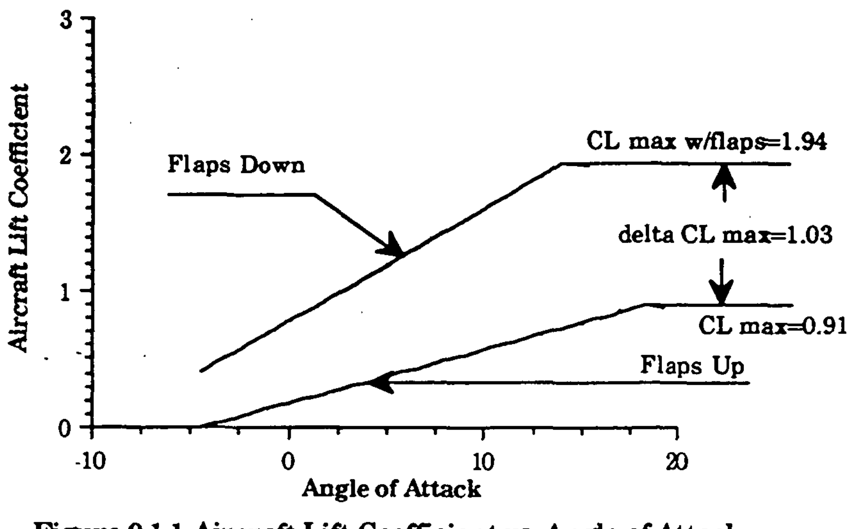


Figure 9.1.1-Aircraft Lift Coefficient vs. Angle of Attack

9.2 DRAG PREDICTIONS

The drag of the aircraft was predicited at various possible flight conditions. The varying parameters included altitude and velocity. The altitude was varied to take into account the effect of air density on Mach number and Reynolds number. The altitude was varied from sea-level to 40,000 feet, and the velocity was varied from 250 to 500 knots. The values for skin friction, parasite drag, and drag due to lift were determined at each condition[17]. Wave drag was also examined and taken into account in the transonic region, but the effect of this is minimal in the flight regime of the aircraft.

Figure 9.2.1 is a plot of lift coefficient versus drag coefficient at cruise conditions. From the graph, the zero lift drag coefficient of the aircraft

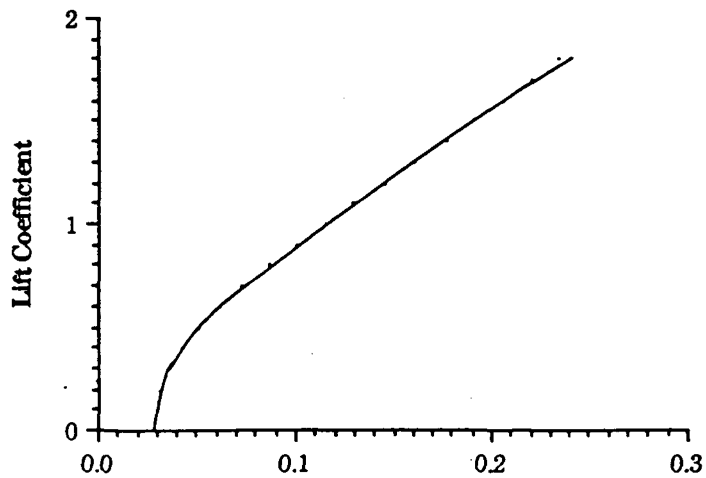


Figure 9.2.1-Drag Polar at M=0.6 and Altitude=20,000 ft.

is approximately 0.03. As mentioned earlier, this value includes the skin friction and parasite drag coefficients. As lift occurs, the drag coefficient increases accordingly. This is due to the increase in drag due to lift. The wing efficiency factor used in this calculation was estimated to be 0.672. In fact, as the lift is increased, the drag due to lift becomes the dominant form of drag for the aircraft.

Figure 9.2.2 shows the predicted zero-lift drag coefficients at various Mach numbers at 20,000 feet. This altitude was determined to be the optimum cruise altitude because of the predicted thrust specific fuel consumption. As can be seen in the figure, the zero-lift drag coefficient is approximately constant up until a Mach number of 0.8. At this point, the zero

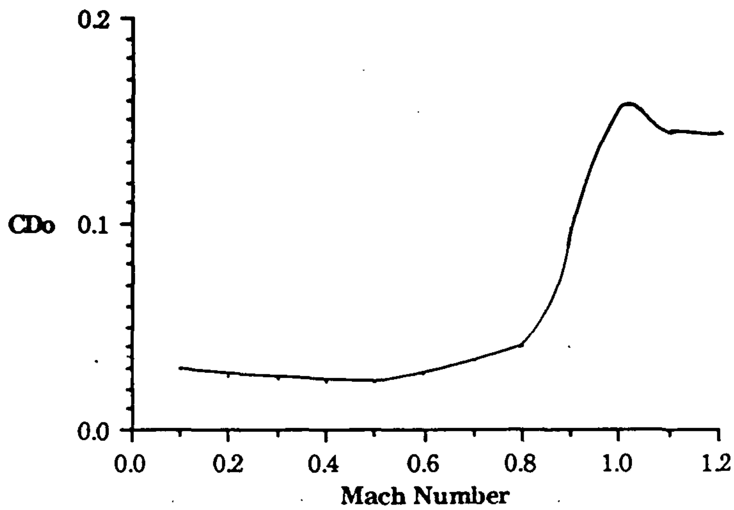


Figure 9.2.2-Zero Lift Drag Coefficient vs. Mach Number

lift drag coefficient rises rapidly due to the wave drag associated with the transonic region. This rapid drag rise above a Mach number of 0.8 is not critical to the performance of the aircraft, though, because the maximum speed of the aircraft is only 0.756.

Figure 9.2.3 is a plot of the predicited Lift/Drag values versus Mach number at the cruise altitude. The predicited L/D values peak at a Mach number approximately equal to 0.5. This would seem to be the optimum cruise Mach number. However, as mentioned earlier, the thrust specific fuel consumption of the aircraft was best at 20,000 feet and at a Mach number equal to 0.6. This shows the trade-offs required to properly design the aircraft. Another example of this is the use of a supercritical airfoil to prevent drag divergence. A standard airfoil would have provided more lift, but the drag rise at higher Mach numbers was deemed unacceptable[11].

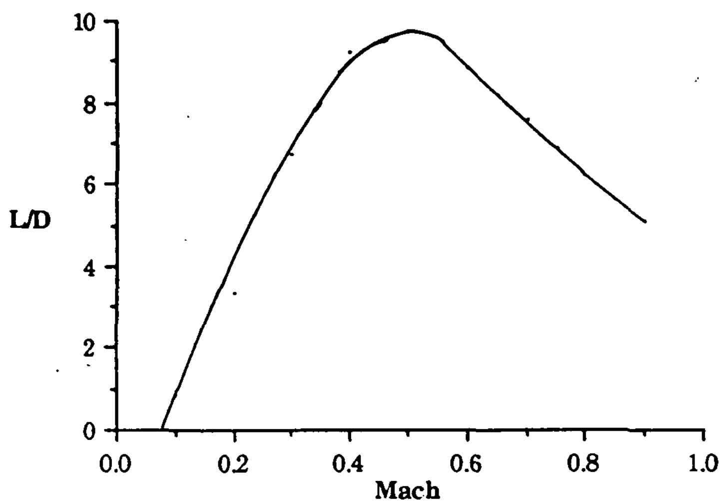


Figure 9.2.3-L/D vs. Mach Number at 20,000 ft

10.0 STABILITY AND CONTROL/HANDLING QUALITIES

A preliminary stability and control analysis for the Cyclone is considered for both static and dynamic flight conditions of the aircraft at three different flight conditions, 1) dash at sea level at Mach 0.756, 2) combat at sea level at Mach 0.684, 3) cruise at an altitude of 20,000 ft at Mach 0.6. After a thorough investigation of the proper sizing of the aircraft, the stability and control analysis indicates that the handling qualities of the aircraft meet level 2 flying conditions as specified in MIL-F-8785B [18] for category A aircraft. A level 2 flying quality identifies an airplane that can be controlled safely; however, there is some increase in the pilot's workload, and/or there is degradation in the mission effectiveness. Therefore, it is deemed necessary to provide an augmentation system for the Cyclone in order to insure that the aircraft will perform its intended mission.

10.1 STATIC MARGIN ASSESSMENT

A longitudinal X-plot as shown in Figure 10.2.1 was generated to aid in sizing the vertical tail of the aircraft.

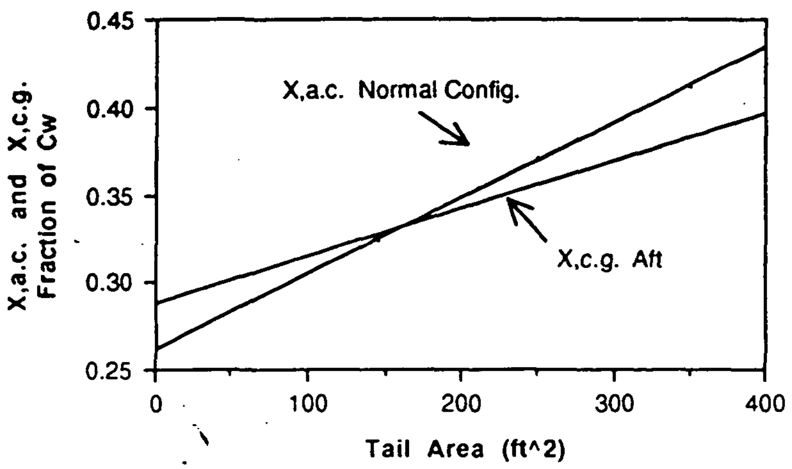


Figure 10.2.1 Longitudinal X-plot

It was decided to design the aircraft to be slightly unstable for the purpose of having manueverability.[3] Too extreme of an unstable design was deemed unsatisfactory do to the inability of the pilot having control of the aircraft if the flight computer was ever to fail. In order to determine the point at which the aircraft would be statically unstable, a projected horizontal tail area of 145 sq. ft. was chosen. From this a static instability of approximately 2% occurred at a gross take-off weight of 54,527 lbf.

10.2 STATIC STABILITY ASSESSMENT

The static stability derivatives for the Cyclone were generated for the three flight conditions outlined in Table 10.2.1 using standard stability derivative equations as defined in DATCOM.[10]

Table 10.2.1 Flight Conditions for Stability Analysis

	Flight Condition 1	Flight Condition 2	Flight Condition 3
Mach	0.756	0.684	0.6
Altitude	Sea Level	Sea Level	20,000 ft
Flight Phase	Dash	Combat	Cruise
Configuration	Fully Loaded	Fully Loaded	Fully Loaded
Static Margin	2%	2%	2%
Weight	54,527 lbf	54,527 lbf	54,527 lbf
I_{XX} (slug-ft^2)	67,518	67,518	67,518
I_{YY} (slug-ft^2)	188,034	188,034	188,034
I_{ZZ} (slug-ft^2)	246,240	246,240	246,240
I_{XZ} (slug-ft^2)	-9085	-9085	-9085

With the static stability derivatives calculated and presented in Table 10.2.2, it is possible to generate dynamic stability relationships from these values as

presented in section 10.4 Literal Factors Determination.

Table 10.2.2 Static Stability Derivatives

Stability Derivative	Flight Condition 1	Flight Condition 2	Flight Condition 3
C_{L_u}	0.124	0.101	0.188
C_{m_u}	0	0	0
C_{D_u}	0.001	0.005	.010
C_{L_α}	2.630	2.520	2.440
C_{m_α}	-0.491	-0.476	-0.460
C_{D_α}	0.027	1.489	0.023
C_{l_β}	-0.0274	-0.045	-0.050
C_{n_β}	0.071	0.071	0.071
C_{y_β}	-0.530	-0.530	-0.530
$C_{L_{\alpha d}}$	0	0	0
$C_{m_{\alpha d}}$	-2.067	-2.067	-2.067
C_{L_q}	0	0	0
C_{m_q}	-2.235	-3.146	-2.872
C_{l_r}	0.390	0.390	0.390
C_{n_r}	-0.187	-0.187	-0.187
C_{y_r}	0.233	0.233	0.233
C_{l_p}	-0..627	-0..609	-0.596
C_{n_p}	0.015	0.016	0.016
C_{y_p}	-4.170	-4.170	-4.170

10.3 CONTROL POWER

Further analysis involved determining static stability control derivatives as shown in Table 10.3.1 for the three flight conditions previously defined. From this analysis the control surface sizing was determined. The initial sizing proved

successful in placing the aircraft within the RFP requirements for the performance analysis as discussed previously in section 3.0 Performance.

Table 10.3.1 Control Derivatives at Flight Conditions

Stability Derivative	Flight Condition 1	Flight Condition 2	Flight Condition 3
$C_{z_{\delta e}}$	0	0	0
$C_{m_{\delta e}}$	-5.674	-5.107	-4.630
$C_{l_{\delta a}}$	0.080	0.080	0.080
$C_{n_{\delta a}}$	-0.003	-0.003	-0.003
$C_{y_{\delta r}}$	-0.251	-0.251	-0.251
$C_{l_{\delta r}}$	0.280	0.280	0.280
$C_{n_{\delta r}}$	-0.251	-0.251	-0.251

10.4 LITERAL FACTORS DETERMINATION

According to the longitudinal and lateral modes of analysis, it was shown in Table 10.4.1 that the Cyclone does not meet level 1 flying quality requirements for both longitudinal and lateral stability. From the prelimainary analysis it was shown that Flight Condition 1 met level 2 flying qualities with a short period damping ratio of 0.268, a phugoid damping ratio of 0.137 and a Dutch roll damping ratio of 0.180. Similarly, Flight Condition 2 met level 2 flying qualities with a short period damping ratio of 0.295, a phugoid damping ratio of 0.131 and a Dutch roll damping ratio of 0.180, and, lastly, Flight Condition 3, also, met level 2 flying qualities with a short period damping ratio of 0.213, a phugoid damping ratio of 0.112 and a Dutch roll damping ratio of 0.132. Therefore, from these dynamic characteristics it is apparent that the Cyclone will be outfitted with a stability augmentation system for in flight control correction.

Table 10.4.1 Level 1 Requirements with Dynamic Stability Analysis

FLIGHT CONDITION 1		
Longitudinal Flying Qualities	**Requirement**	**Unaugmented**
Short Period Damping Ratio	0.35 to 1.30	0.268
Short Period Frequency (rad/sec)	1.5 to 9.0	4.557
Phugoid Damping Ratio	0.04 (minimum)	0.137
Lateral Flying Qualities		
Dutch Roll Frequency (rad/sec)	1.0 (minimum)	3.220
Dutch Roll Damping Ratio	0.19 (minimum)	0.180
Roll Time Constant (sec)	1.4 (maximum)	0.087
FLIGHT CONDITION 2		
Longitudinal Flying Qualities	**Requirement**	**Unaugmented**
Short Period Damping Ratio	0.35 to 1.30	0.295
Short Period Frequency (rad/sec)	1.5 to 9.0	4.065
Phugoid Damping Ratio	0.04 (minimum)	0.131
Lateral Flying Qualities		
Dutch Roll Frequency (rad/sec)	1.0 (minimum)	2.896
Dutch Roll Damping Ratio	0.19 (minimum)	0.180
Roll Time Constant (sec)	1.4 (maximum)	0.100
FLIGHT CONDITION 3		
Longitudinal Flying Qualities	**Requirement**	**Unaugmented**
Short Period Damping Ratio	0.35 to 1.30	0.213
Short Period Frequency (rad/sec)	1.5 to 9.0	2.363
Phugoid Damping Ratio	0.04 (minimum)	0.112
Lateral Flying Qualities		
Dutch Roll Frequency (rad/sec)	1.0 (minimum)	1.728
Dutch Roll Damping Ratio	0.19 (minimum)	0.132
Roll Time Constant (sec)	1.4 (maximum)	0.233

11.0 AVIONICS

11.1 AVIONICS PHISOSOPHY

The philosophy involved in the selection of necessary airborne equipment is straight-forward. The close air support aircraft must be highly capable as a weapons delivery vehicle during the day, night or in adverse weather conditions. It must also be survivable in the presence of hostile threats. Every possible means should be sought to accomplish these ends.

Former close air support aircraft, such as the Fairchild A-10 were designed to operate by day, under the presumption that an aircraft with simple systems would prove more rugged and capable than its complex supersonic counterparts[1]. While this may still be the case in some contemporary situations, it is fairly clear in light of recent developments that the rules of the game have changed. The majority of close air support missions flown during Desert Storm operations took place under the cover of darkness in order to deny the enemy visual perception of movement and preserve the surprise advantage. Aircraft possessing both day and night capability proved highly successful in carrying out their missions, experiencing remarkably few losses. The Cyclone will possess such capability.

Simplicity of design is unmistakably an advantage in a battlefield situation, however, it must not be forgotten that simplicity also applies to operation of the aircraft. The workload demanded of the pilot in the foreseeable future air-ground scenario will be overwhelming without the

aid of pilot assistance systems. The importance of an advanced cockpit cannot be more highly stressed. The Cyclone aircraft will carry all the necessary systems to carry out its mission.

The requirement of a capable navigation system is an integral part of any military aircraft. However, in the case of the close air support role this becomes indispensable. It must not be assumed that the future aircraft will be able to return to the same airfield it departs from. For this reason it must be able to navigate, independent of ground aids, at night and in adverse weather in unfamiliar territories. In light of this consideration, both an INS and terrain following radar will be incorporated in the Cyclone. The advent of the ring laser inertial navigation system has significantly reduced the maintenance time associated with conventional mechanical units due to the reduction of moving parts. This characteristic will be incorporated into the Cyclone wherever possible to reduce down time and increase system dependability.

11.1 AVIONICS SYSTEMS

The following description of the avionics system is based on current proven technology. In the event of technology up-grades, substitutions will be made, however the described systems are not expected to change significantly.

The avionics system of the Cyclone will be made of a number of separate modular components all linked through two Control Data Corp AYK-14 mission control computers by means of a MIL-STD 1553B digital data buss. In order to facilitate day, night and adverse weather operations, two Martin Marietta LANTRIN (low altitude and targeting infra-red for night system) pods will be incorporated. The navigation pod will include a

wide field-of-view forward infra-red unit, terrain following radar, power supply, pod control computer and an environmental control unit. The targeting pod will include a stabilization system, wide and narrow field laser designator/ranger, automatic target recognizer, automatic Maverick hand-off system, power supply and environmental control unit. Although the cost of the LANTRIN system is high in comparison to standard FLIR systems, the added capability of terrain following radar and a laser designator justify the expense. Due to the high cost of the LANTIN system, all 500 aircraft will be outfitted with external fuselage pylons to accept the pods, but only 50 pods will be acquired.

The selection of the nose mounted radar reflect the intended mission of the Cyclone. Basically an air to ground fighter, the Cyclone will have little use for a large, high pulse repetition radar (PRF). High PRF radars, while offering more range in the air-to-air mode, are impractical in the look down mode or attack mode due to scattering from ground clutter. With this in mind a smaller, more closely suited unit such as the Westinghouse night/adverse weather attack radar will be used, thus saving weight, energy and space.

The navigation system will include an SKN-4030 ring laser gyro INS coupled with a standard TACAN unit. The INS will be updated periodically with information from the LANTRIN pod.

Electronic counter measures will be addressed by an ALR-67 radar warning receiver by Itek and an ALQ-165 radar jammer by ITT/Westinghouse. An ALE-40 flare and chaff dispenser by Tracor will be used to counter infra-red and radar missile threats.

The cockpit will be advanced, utilizing two multi-function displays by Kaiser Aerospace and a horizontal situation display by Bendix. The HUD

will be a wide angle GEC unit compatible with the LANTIN system. The hands on throttle and stick (HOTAS) concept will be used, allowing the pilot to maneuver and change HUD modes without moving his hands from the controls. All weapons and stores will be managed by the mission computers and easily displayed on a multi-function display at the pilot's command.

In the unlikely event of a total power failure, redundant critical flight instruments will be carried to supplement the CRT and HUD displays. These will include mechanical airspeed, vertical speed, altitude, attitude and fuel quantity gages.

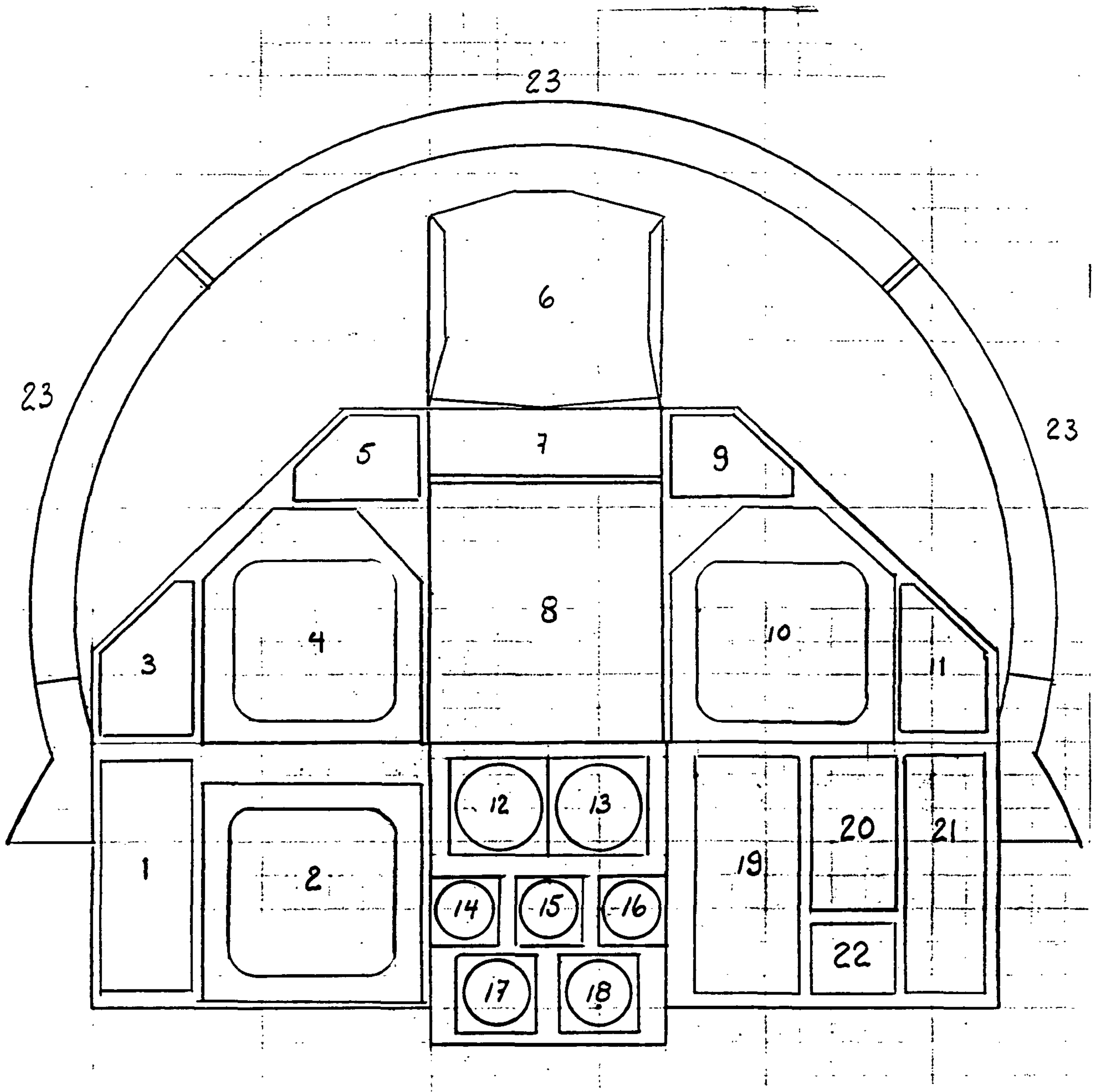


FIGURE 11.1.1 Cyclone Cockpit Instrumentation

1. ECM control panel
2. Horizontal situation display
3. Master armament panel
4. Master monitor display
5. Left warning panel
6. Head-up display
7. HUD camera
8. Up-front control panel
9 Right warning panel
10. Multi function display
11. Caution light panel
12. Stand-by attitude reference indicator
13. Radar warning display
14. Stand-by airspeed indicator
15. Stand-by altimeter
16. Vertical airspeed indicator
17. Clock
18. Cabin pressure altimeter
19. Digital engine display
20. Fuel quantity indicator
21. Stores jettison indicators
22. Landing gear panel
23. Mirror

12.0 SYSTEM LAYOUT

The hydraulic, electrical and fuel systems of the Cyclone are designed to be as redundant and reliable as possible. Maintainability and accessibility is also a prime concern in the preliminary design phase.

12.1 HYDRAULIC SYSTEM

The Cyclone includes three Hydraulic systems. The basic system components are pictured in Figure 12.1.1.. The number 1 and 2 hydraulic pumps are driven by the engine accessory gearbox, each pump will be designed to supply both systems with adequate volumetric flow rate. The auxiliary, electrically operated hydraulic system purpose is two fold. First it supplies hydraulic power to the flight controls in the unlikely event of the other hydraulic system failure. Secondly the hydraulic system can be energized on the ground by the maintenance crew without the need for expensive and difficult to transport hydraulic power carts.[19]

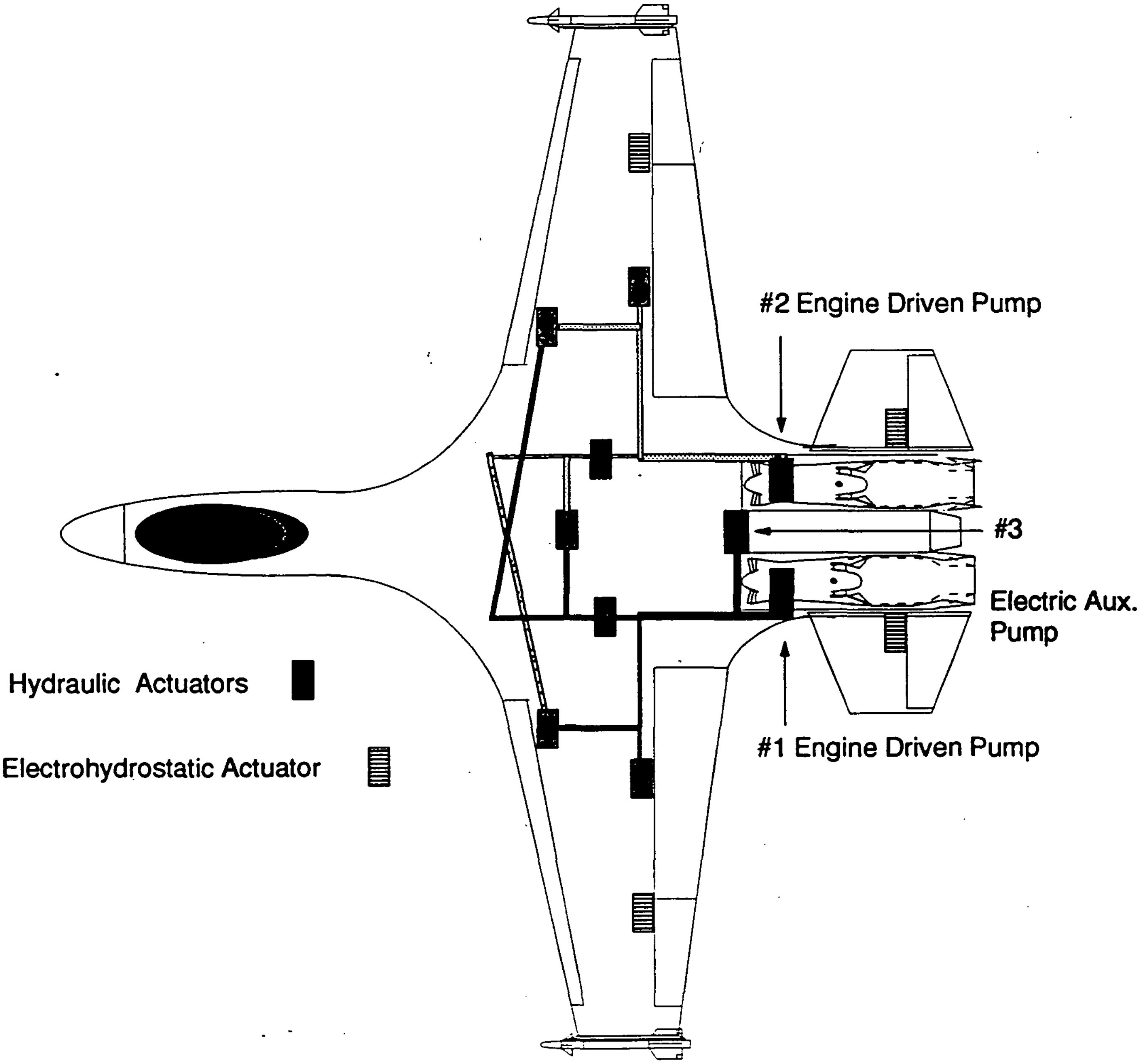


Figure 12.1.1 Hydraulic System Layout

12.2 ELECRICAL SYSTEM

The basic power system design principle is pictured in Figure 12.2.1. This figure depicts the redundant system philosophy of the Cyclone. The aircraft electrical components generally have a dual power supply (access to each generator). When this is not possible the components themselves are doubled. Hence, should one power system fail all electric system functions are ensured.

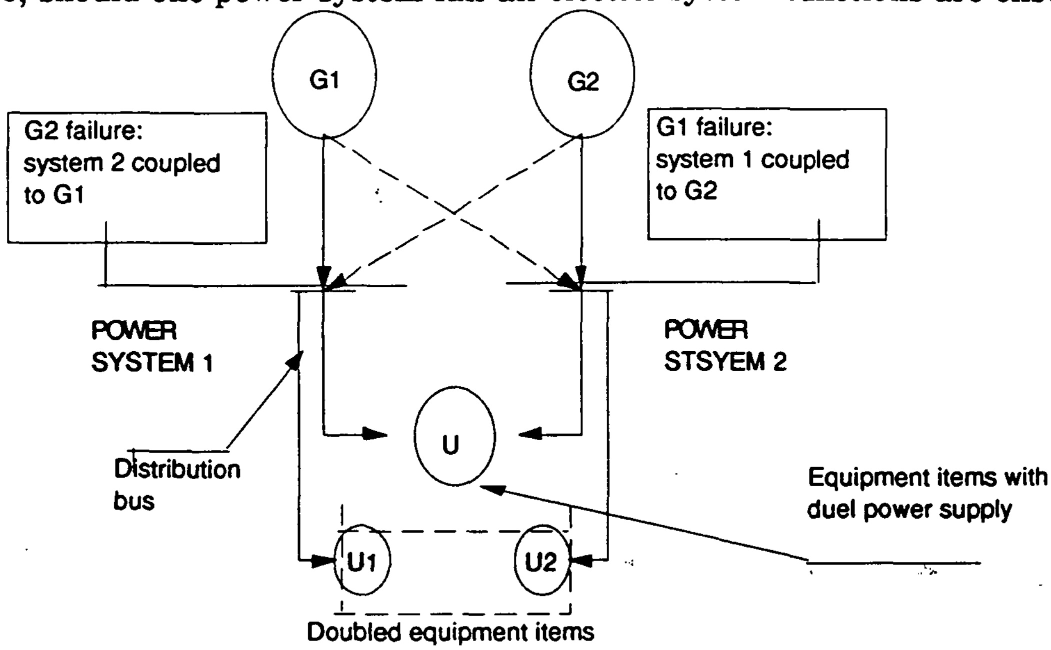


Figure 12.2.1 Elecrical Generation Schematic

The electrical generation system of the Cyclone contain four main components, these include: Two generators each driven by the engine accessory gear box, a self contained auxiliary power unit (APU) used to start the engines and enable the ground crew to operate all the electrical systems on the ground without a difficult to transport and expensive ground power unit. Since a fly- by -wire flight control system will be used on the Cyclone a battery located in the nose area of sufficient size to operate the primary safety of flight components for approximately 45 min. is incorporated in the unlikely event of a total electrical system failure.

the primary safety of flight components for approximately 45 min. is incorporated in the unlikely event of a total electrical system failure.

12.3 FLIGHT CONTROL SYSTEM

The Cyclone will incorporate a Fly-by-wire flight control system due to its unstable design. The primary advantages of this type of control system over the conventional hydraulic system are ease of weapon's integration, and reduction of the pilot work load in a combat situation, readily adapted to control mixing and Weight savings.[4] The primary control surface actuators will utilize electrohydrostatic actuators. These actuators have their own miniature hydraulic system and offer weight advantages as well as being easily integrated into a fly by wire system.[4]

12.4 FUEL SYSTEM

The Cyclone fuel is contained in six self-sealing bladder tanks within the fuselage. The fuel system incorporates transfer pumps, boost pumps, venting system and the necessary fuel dump components. The Cyclone is also equiped with an in flight refueling receptacle to extend the range and to enable the aircraft to stay near the battlefield without returning to refuel. The Cyclone has four main fuel tanks that feed into two main supply tanks located between the aft wing bulkhead and the forward engine support structure. Each engine can be operated from its own supply tank or if one tank is damaged, both engines can be operated from one. A preliminary fuel system design is pictured in Figure 12.4.1 which shows the principle components.

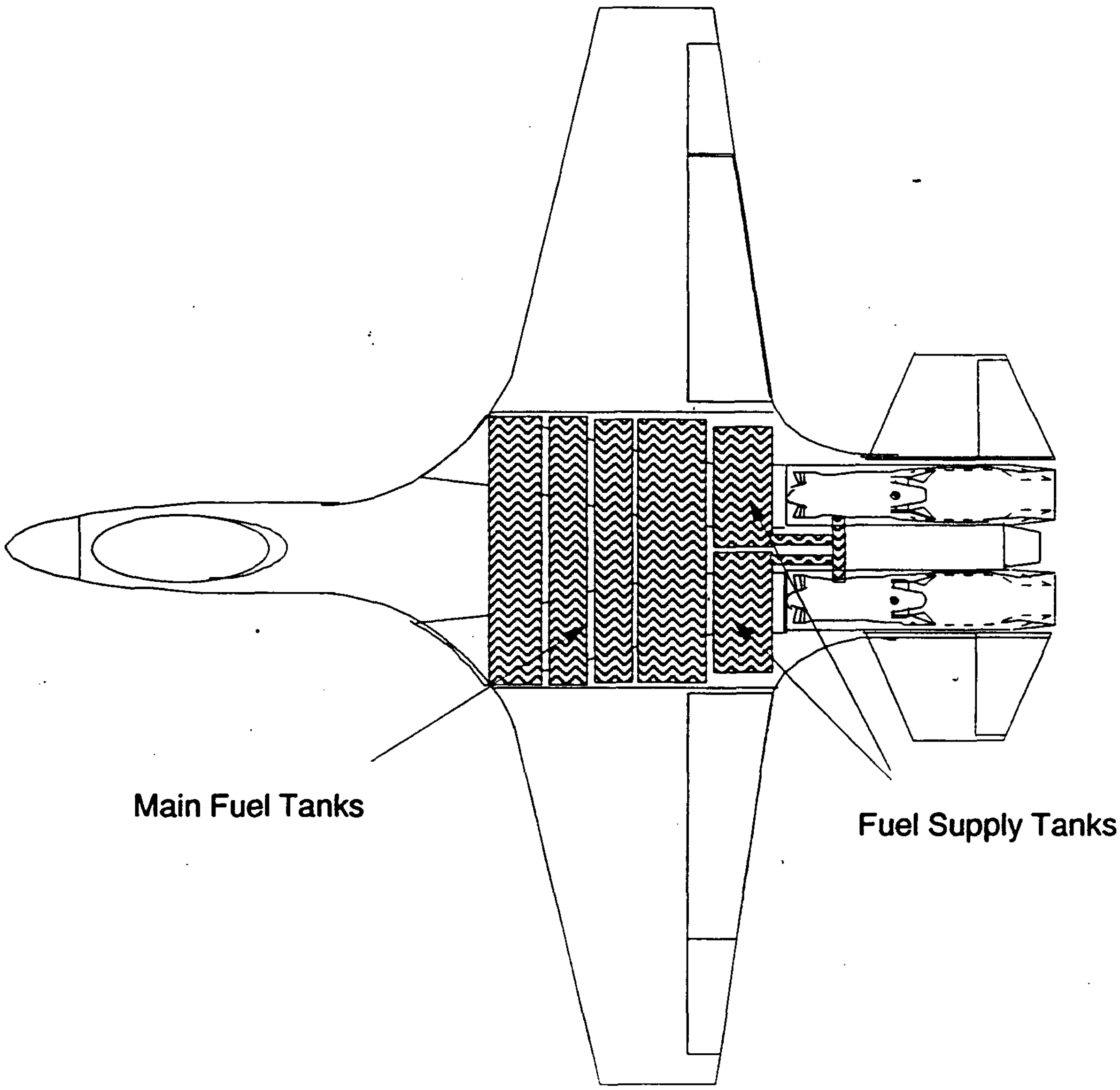


Figure 12.4.1 Fuel System Layout

13.0 WEAPONS INTEGRATION

The Cyclone attack aircraft is equipped with seven external stores locations, all of which may be occupied simultaneously. The centerline pylon, when fitted, would normally carry an external fuel tank. Because of ground clearance restrictions, it may be fitted only when a hard surface runway is available. Two more pylons under the fuselage forward of the main gear, and the four pylons under the wings are available to carry a full range of guided and unguided weapons and pods. Each is rated at 3500 pounds, enough to support a standard multiple ejector rack with six 500 pound bombs. Two additional attach points are provided under the forward fuselage on either side of the nose gear location, as indicated in Figure 13.1. These are suitable to carry specialized avionics packages, such as the LANTIRN navigation or targeting pods, as the mission requires.

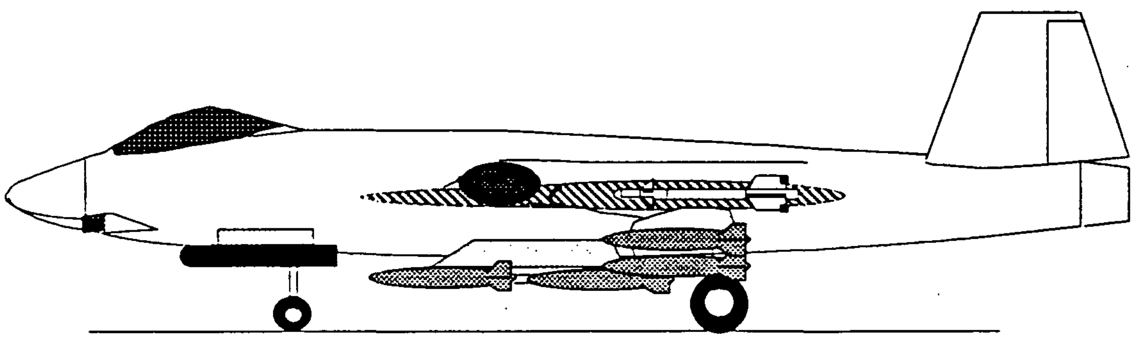


Figure 13.1. Placement of port avionics pod

Various mission loads for Cyclone are diagrammed in Figure 13.2. The design attack mission requires carriage of twenty Mk 82 free-fall bombs. This is achieved by placing a triple ejector at each of the underwing pylons, and a multiple ejector rack with four bombs at each ventral pylon. The maximum of

thirty-two bombs of the 500-pound class can be carried when multiple racks are used on all six pylons and when either short-field performance (in the form of excess lift), range (fuel) or maneuverability (g-loading) can be sacrificed in exchange for the extra ordnance.

In most combat situations a far smaller load would be carried, with the advantages of range, loiter time, and performance. Most of the loadings shown in the figure represent less than half the weight of the design payload of 10,000 pounds. For some roles, such as combat rescue escort or forward air control, external fuel is used to further increase loiter time. For long range ferry flights, additional fuel pods can be fitted under the wings to augment internal fuel capacity by over sixty percent.

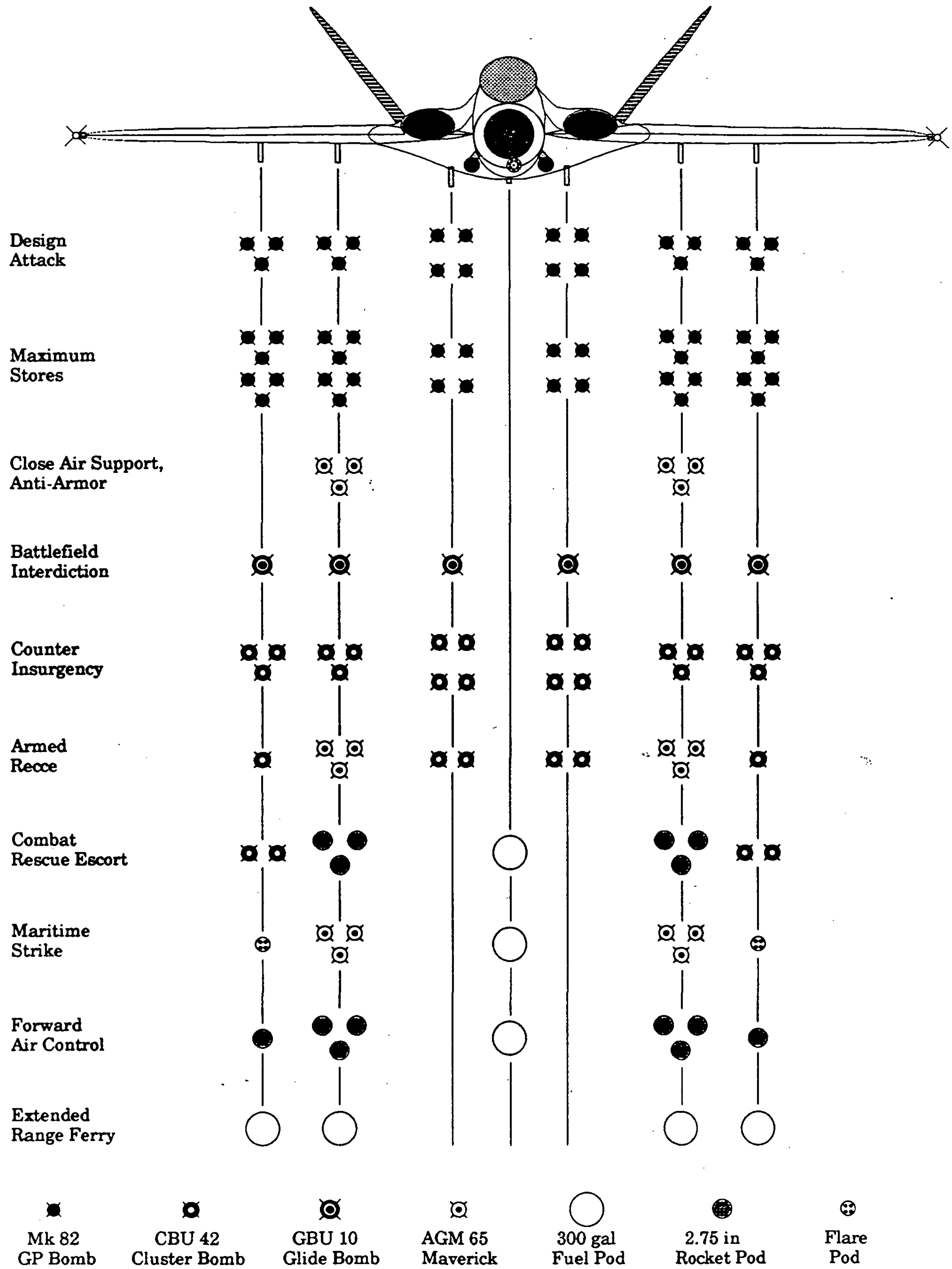


Figure 13.2. Sample mission payload configurations

14.0 GROUND SUPPORT REQUIREMENTS

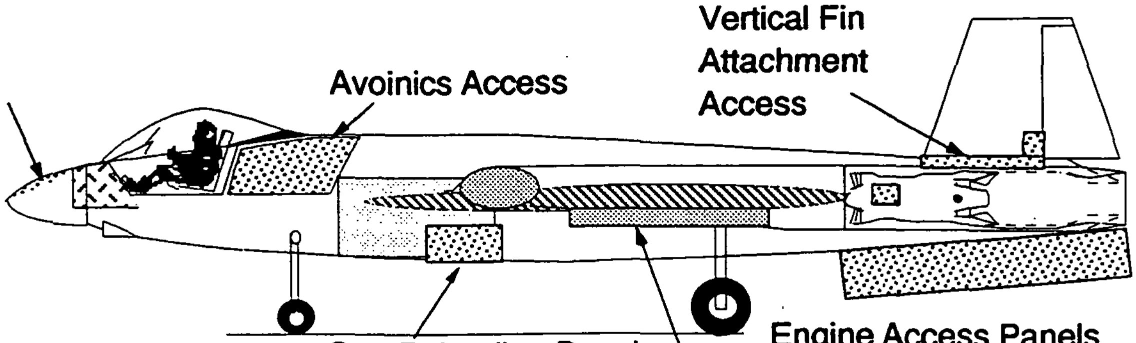
14.1 FLIGHT SERVICE REQUIREMENTS

The Cyclone was designed to minimize special flight service equipment . Because the Cyclone's cannon is identical to the cannon in the A-10 aircraft the currently used gun- reloading carts can be utilized, reducing the support equipment cost. Similarly, the existing bomb carts can also be used on the Cyclone. Refueling of the aircraft is accomplished by one single point fueling resceptacle located on the fuselage below and slightly behind the wing leading edge. This position was chosen so that the ground support personal can perform "hot refueling" operations and be out of the inlet suction danger zone.

14.2 MAINTENANCE REQUIREMENTS

The Cyclone deemed repairability and accessibility important in the over all success of this aircraft in a battle situation. The major systems that require fluid level reservoirs will be equiped with easy to read sight gauges reducing the on-ground turn around time. The majority of the aircraft system components will be attempted to be placed low on the fuselage to alleviate the need for the maintenance personnel to use ladders to gain access for their replacement. Where this is not possible attachment points for fuselage side mounted ladders will be provided. The engine replacement of this type of aircraft is of particular concern. The Cyclone's engines are accessible by a large fold down cowl under each engine see Fig 14.2.1

Inspection Access



Access Panel Locations
Figure 14.2.1

The engine fuel system lines will be removable by quick-disconnect fittings and the wiring harasses by cannon plugs. The engine itself is mounted on typical fixed and free sliding trunnion see Figure 14.2.2 this enables the engines to be slid aft and the lowered onto an engine cart for replacement. Based on similar aircraft with this type of replacement sequence the total engine replacement should take less than 45 min.[1]

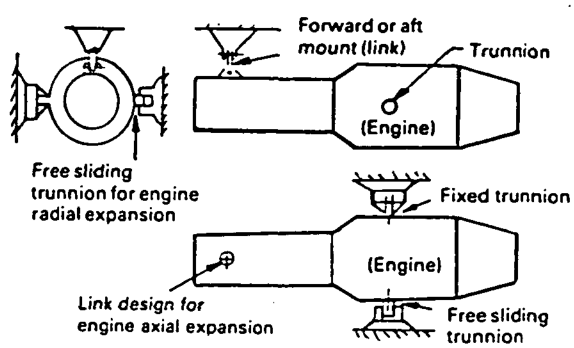


Figure 14.2.2

Reference 16

15.0 COST ANALYSIS

Throughout the evolution of the Cyclone design, it was important to maintain a philosophy of producing a low cost CAS aircraft. With procurement costs of $60 million for one F/A-18 multi-role attack fighter and $75 million for one F-14D multi-role attack fighter[20] (costs based upon the Congressional Budget Office estimates for 1992), it is obvious that the U. S. Congress and Military are seeking lower cost specific role military aircraft. Therefore, a life cycle cost (LCC) analysis was performed on the Cyclone based upon the following four categories:

1) Research, Development, Test and Evaluation (RDT&E) Cost
2) Acquistion Cost
3) Maintenance Cost
4) Disposal Cost

Table 15.0.1 shows the LCC estimate which is based upon a Defense Contractor's Planning Report [21] (DCPR) empty weight of 16,535 lbs, a maximum velocity of 500 knots and a production quantity of 500 aircraft as specified in the RFP.

Table 15.0.1 Life Cycle Analysis

RDT&E	$662.29 million
Acquistion Cost	$8,317.54 million
Maintenance Cost	$13,700.0 million
Disposal Cost	$226.8 million
Total	$22.9 billion

15.1 RDT&E COST

The RDT&E cost for the Cyclone was estimated based upon the production of eight aircraft which are considered independent from the 500 to be produced.

Further, Table 15.1.1 shows the breakdown of the RDT&E cost for the Cyclone with the understanding that RDT&E costs were based upon standard manufacturing costs from previous military aircraft programs[21]. An exception to this estimate involves the addition to the RDT&E cost in modifying an already "off-the-shelf" propulsion system.

Table 15.1.1 Cyclone RDT&E Cost Breakdown

RDT&E Cost Breakdown	Dollars in Millions
Airframe Engineering and Design Cost	92.58
Development Support and Testing Cost	32.73
Flight Test Cost	38.62
Flight Test Operations Cost	12.08
Test and Simulations Facilities	0
RDT&E Profit	52.36
RDT&E Finance Cost	86.39
Subtotal	662.29

15.2 ACQUISITION COST

The Cyclone's acquisition cost is broken down into five areas: airframe engineering and design, airplane program production, production flight test operations, manufacturing finance and profit acquired. Further, to keep the Cyclone production cost even lower, it was decided that for every ten aircraft manufactured only one LANTIRN navigation and targeting system would be delivered. Therefore, with only 50 LANTIRN systems at a cost of approximately $10 million each[5], a major savings of $4.5 billion is achieved for the total acquisition

cost. Table 15.2.1 shows the breakdown of the acquisition cost based upon 1991 dollar amounts.

Table 15.2.1 Cyclone Acquisition Cost Breakdown

Acquisition Cost Breakdown	Dollars in Millions
Airframe Engineering and Design Cost	105.32
Program Production Cost	
Engines	3500.0
Avionics	1350.0
Manufacturing	868.55
Manufacturing Materials	602.82
Tooling	111.07
Quality Control	112.91
Subtotal	6545.36
Production Flight Test Operations	223.32
Total Cost	6873.99
10% Profit Margin	756.14
Total Acquisition Cost	8317.54
Unit Cost (1991 Dollars)	18.02

15.3 OPERATING COST

The program operating cost for the Cyclone is broken down into three main areas: fuel costs, crew salaries and basic maintenance expenses.

The program fuel cost is based upon the most restrictive mission for the Cyclone in which the maximum fuel consumption per aircraft was estimated to be 12,797 lbs at an estimated cost of $0.75 per gallon. In addition, an average mission

time of 1.6 hours at an annual utilization of 300 hours with 409 aircraft in service was estimated in order to determine the number of missions expected each year which was approximately 188. With this information Table 15.2.2 shows the estimated cost of fuel for the Cyclone program.

Table 15.2.2 Cyclone Program Maintenance Cost

	Dollars in Billions
Fuel Cost	2.25
Personnel Salaries	5.20
General Maintenance	6.24
Total	13.7

The personnel salaries for the Cyclone program include those personnel directly and indirectly involved with the operation and maintenance of the aircraft. The Cyclone program is based upon 409 aircraft in service with one pilot at a salary of $38,000 per year and an estimated maintenance of 13 hours per flight hour.[21] Further, a maintenance labor rate of $45 per hour was used, and Table 15.2.2 shows the estimated program cost for all personnel as well as the maintenance. Therefore, the Cyclone operating cost per flight hour is estimated to be $5,583.

15.4 DISPOSAL COST

The disposal cost for a Cyclone aircraft is based upon a 20 year service life of the aircraft.[21] After the service life the disposal will consist of, 1) disassembly of the engines and removal of all electronics, 2) draining of all fluids and their disposal, 3) cutting up of the airframe, 4) salvage of the resulting materials. A disposal cost of $226.8 million was estimated based upon the previous criteria.

16.0 MANUFACTURING BREAKDOWN

16.1 MANUFACTURING FACILITIES

The Cyclone is made of conventional materials and has no radical features that would require major retooling of existing aircraft production assembly lines. Typical manufacturing methods and processes can be used in the production of the Cyclone. The forming and assembly of metal components does not require new technoligies. Since there are no primary aircraft structures made of modern composite materials, complex and expensive autoclave facilities are not required in the manufacturing process. An existing aircraft manufacturing plant could be contracted to produce the Cyclone to avoid constructing a new aircraft manufacturing plant .

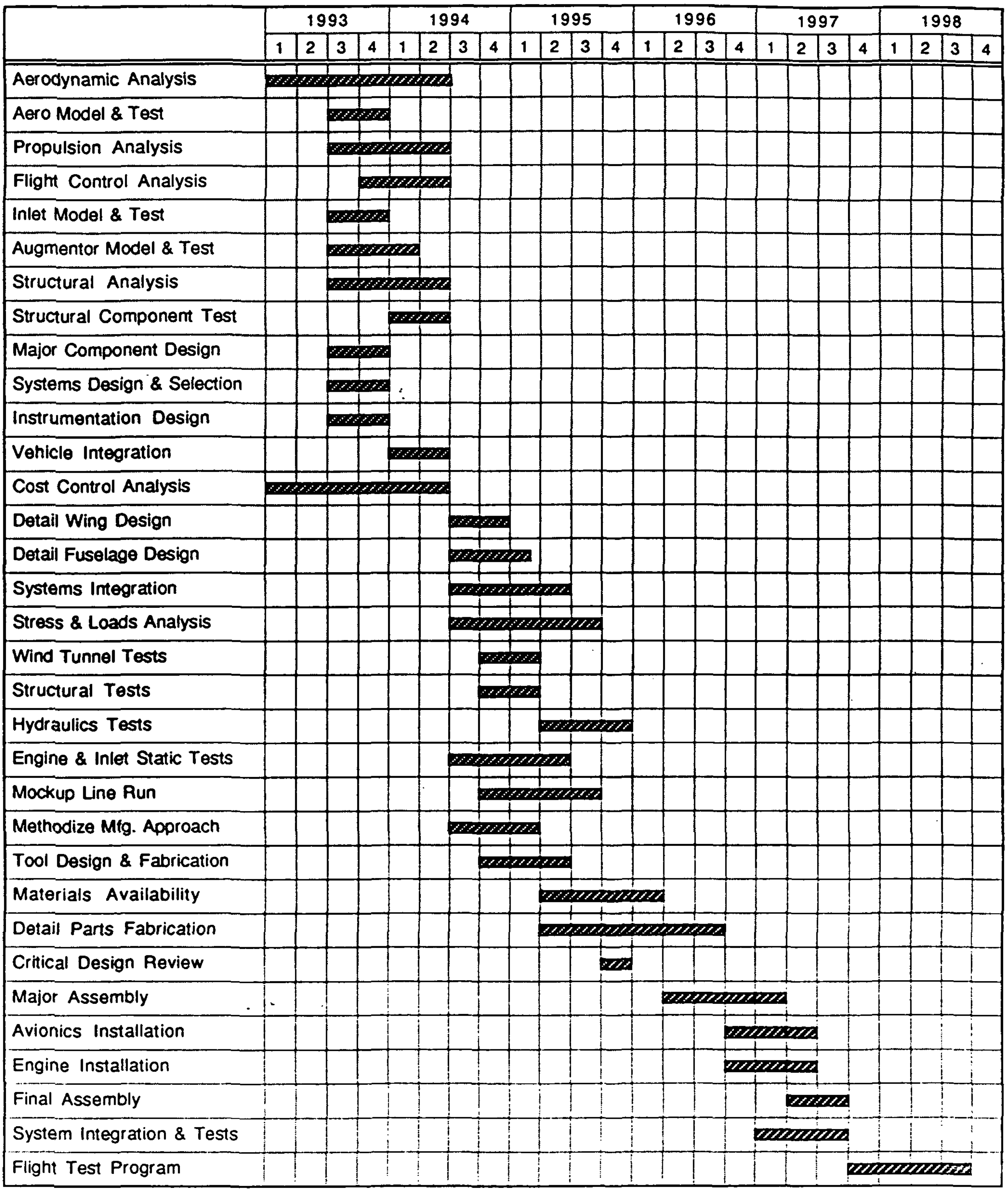
16.2 PRODUCTION SCHEDULE

The production schedule for the Cyclone is outlined in Table 16.2.1. The production time period for the manufacture of 500 copies of the Cyclone is six years. Transition from one production activity to the next is overlapping in some cases to integrate the production and testing processes of several manufacturing activities simultaneously. At the beginning of manufacture, the production speed will be less than the normal production speed due to the learning curve of production personnel and streamlining the flow of materials, tools, hardware, testing, and inspection. Based on a production order of 500 aircraft over a 6 year production run a normal production speed is 20.83 units per quarter.

16.2.1 PRODUCT ASSEMBLY

The major aircraft component assembly sequence of the Cyclone is shown in Figure 16.2.1.1. The assembly of the aircraft in this manner enables the

Table 16.2.1 Cyclone Production Schedule



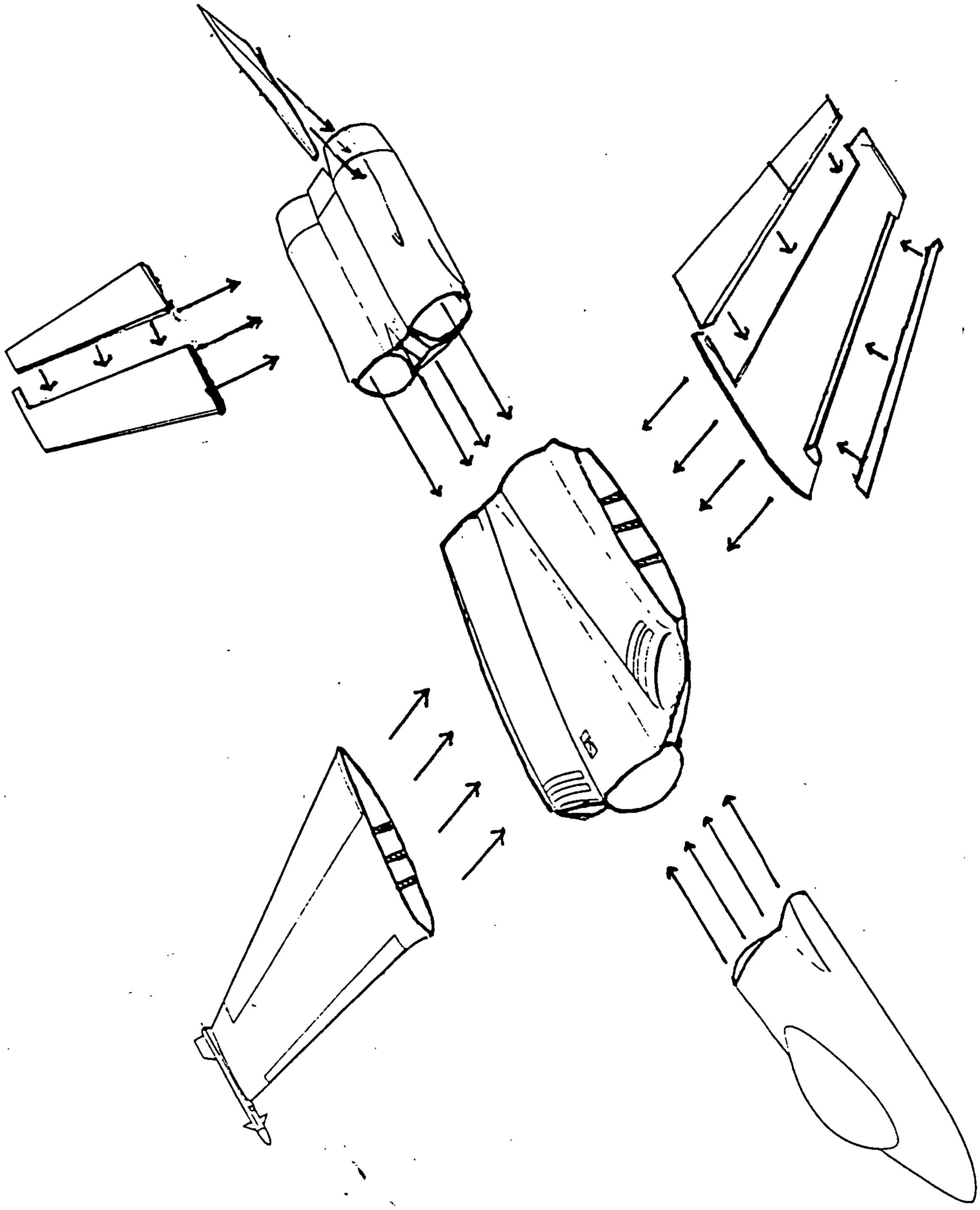


Figure 16.2.1.1 Product Assembly

major sub-assembles to be subcontracted out or assembled in other plants for possible labor savings in other locations and increase production facility utilization.

16.3 MANAGEMENT STRUCTURE

The aircraft management structure is shown in Figure 16.3.1. To ensure that the cost of the Cyclone is maintained within budget constraints the secretary-treasurer is to be involved in all decisions concerning deviations from the manufacturing plan and budget reviews.

17.0 CONCLUSION

The Cyclone meets or exceeds each design parameter that was initially set forth as a goal. To accomplish this, a variety of analytical methods were employed, and the optimum design was arrived at after many design trade-offs were investigated. These trade-offs included choices between engine types and fuselage configurations. Although the chosen turbofan engine doesn't provide the same amount of speed as a turbojet engine, it does have a better specific fuel consumption and fits better into the Cyclone's regime of subsonic speeds.

Overall, it is strongly believed that the aircraft design outlined in this report would readily satisfy the future requirements of the military for a close air support aircraft. The ability to employ the LANTIRN system as a navigation and targeting device makes the Cyclone very adaptable to a variety of warfare scenarios. In combination with the variety of stores that the Cyclone can carry, this makes it a very formidable weapons delivery system.

Before the fruition of this aircraft design, the stability and control should be investigated further in flight conditions other than those investigated in the preliminary design process. Furthermore, data such as that obtained in a wind tunnel would be helpful in verifying stability criteria.

In the future, costs could possibly be reduced by integration of existing powerplants. The General Electric F404 is an engine that could be employed, but data regarding this powerplant would have to be available for proper investigation of its possible use.